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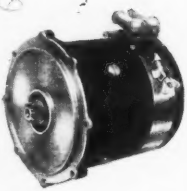

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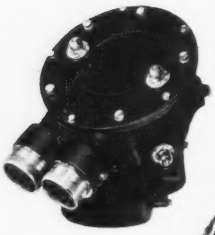

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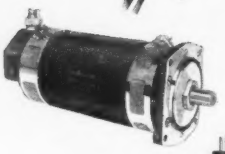

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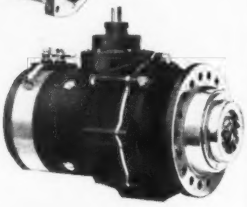

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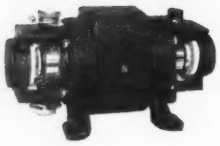

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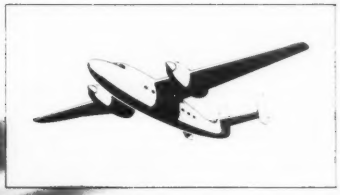

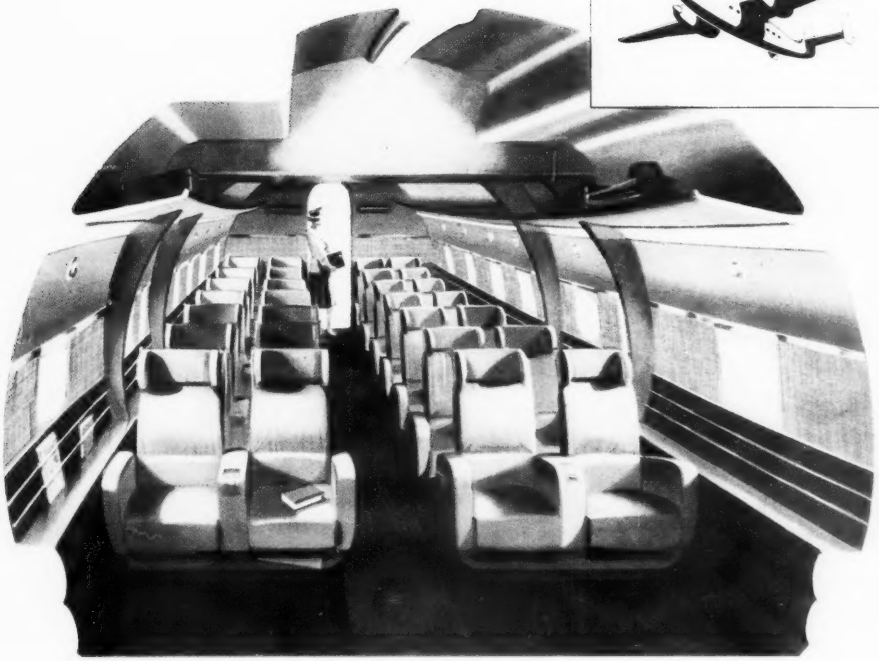

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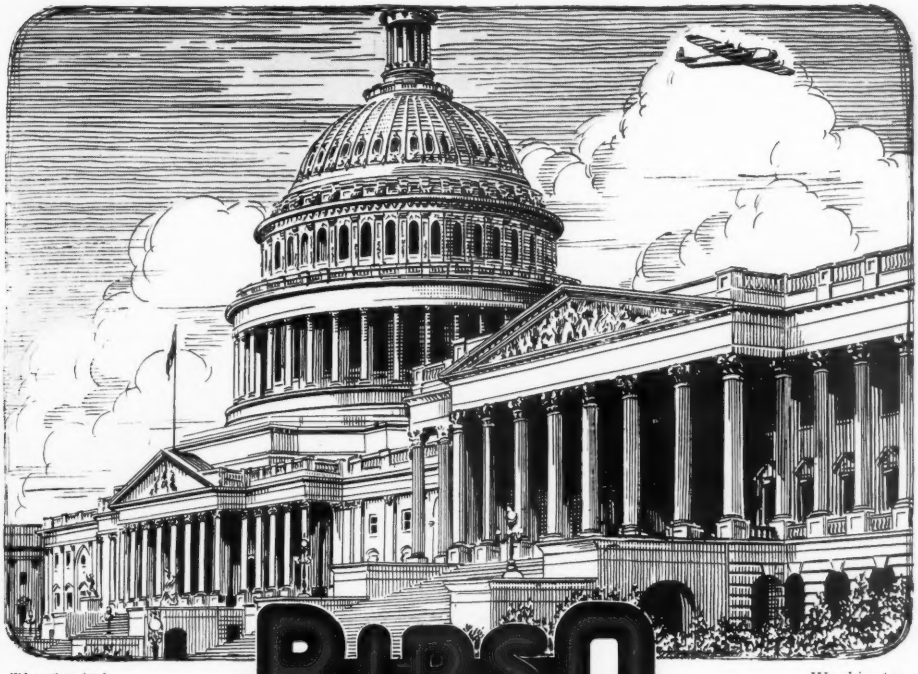

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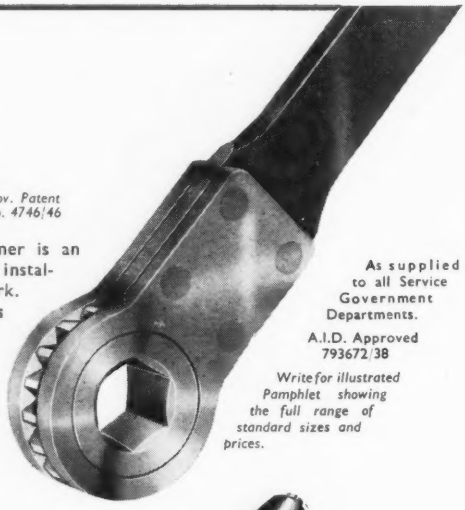

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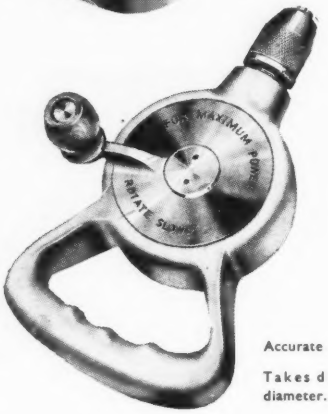

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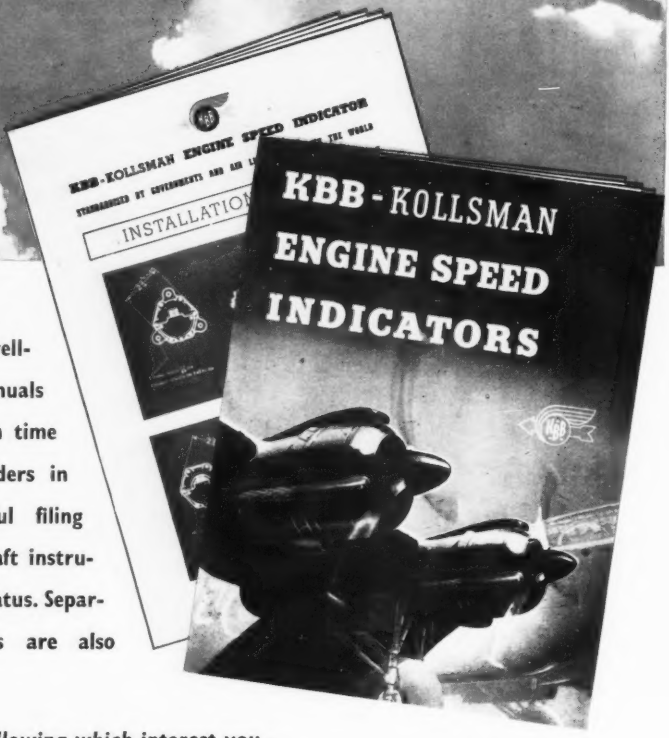

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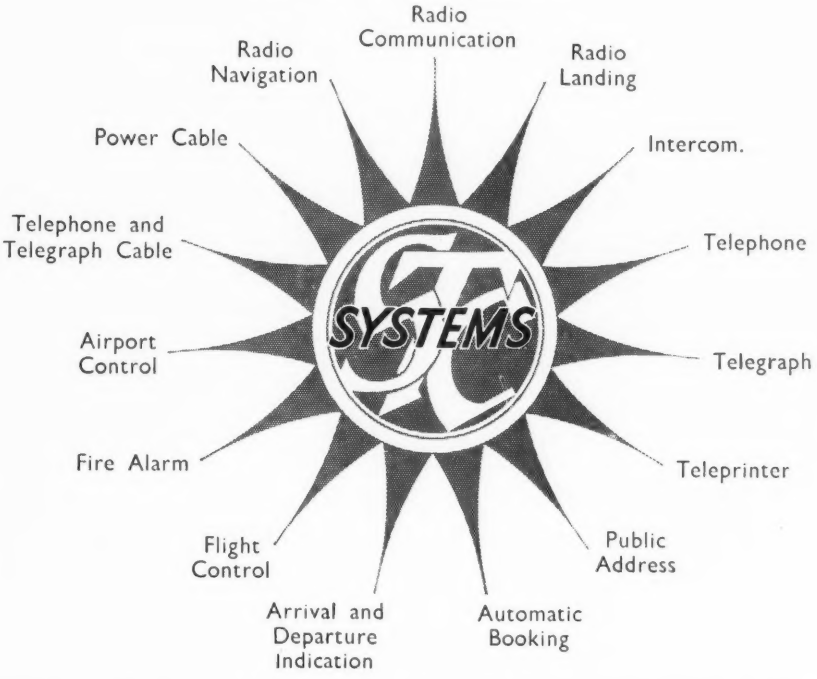

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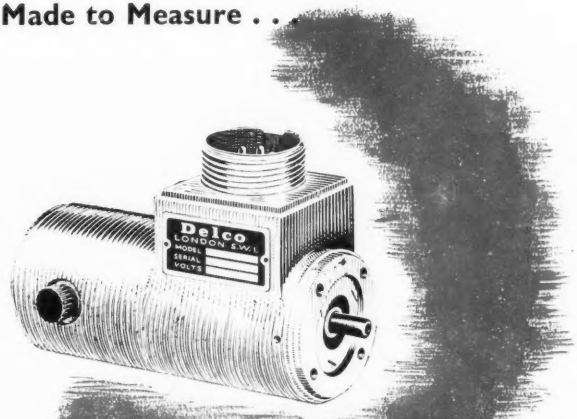

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A CRITICAL REVIEW OF GERMAN LONG-RANGE ROCKET DEVELOPMENT

by

W. G. A. PERRING, F.R.Ae.S.

Mr. Perring trained as a Naval Architect at the Royal Naval College, Greenwich, and did post-graduate research work at the National Physical Laboratory at Teddington. He joined the Aerodynamics Department, Royal Aircraft Establishment, Farnborough, in December 1925 *and was appointed Superintendent, Scientific Research, in* 1940. *Since* 1941 *he has been Deputy Director, Research and Experiments, at the Royal Aircraft Establishment. He is a Fellow of the Society.*

A MEETING of the Royal Aeronautical Society was held in the lecture hall of the Institution of Mechanical Engineers at Storey's Gate, St. James's Park, Westminster, London, S.W.1, on Thursday, 1st November 1945, at which a paper by Mr. W. G. A. Perring, F.R.Ae.S., entitled "A Critical Review of German Long-Range Rocket Development," was presented and discussed. In the Chair, the President, Sir Frederick Handley Page.

The President, introducing Mr. Perring, said that he was trained as a naval architect at the Royal Naval College, Greenwich. He had done post-graduate research work at the National Physical Laboratory, Teddington, as an 1851 post-graduate research worker. That date meant that the money which was devoted to that work had become available from the great Exhibition which was held in 1851; it had not to do with the year in which Mr. Perring had done the work !

Later, Mr. Perring had served as Assistant to Sir George Hunter, of Messrs. Swan, Hunter & Wigham Richardson, Newcastle-on-Tyne. He had joined the Aerodynamics Department of the Royal Aircraft Establishment in December, 1925; was appointed Superintendent of Scientific Research at the R.A.E. in May, 1940 and Deputy Director of Research and Experiments in July, 1941.

Mr. Perring presented his paper.

INTRODUCTION. SCOPE OF GERMAN EFFORT

SERIOUS rocket development was started in Germany in the years 1929-1930 by a few groups of private inventors. This work attracted the attention of the Army Weapons Group in 1933, and in 1937-1938 a special research and development station was set up at Peenemunde at the cost of 300 million marks, and the work was transferred from Berlin to this new station at about that time.

Peenemunde concentrated mainly on bi-

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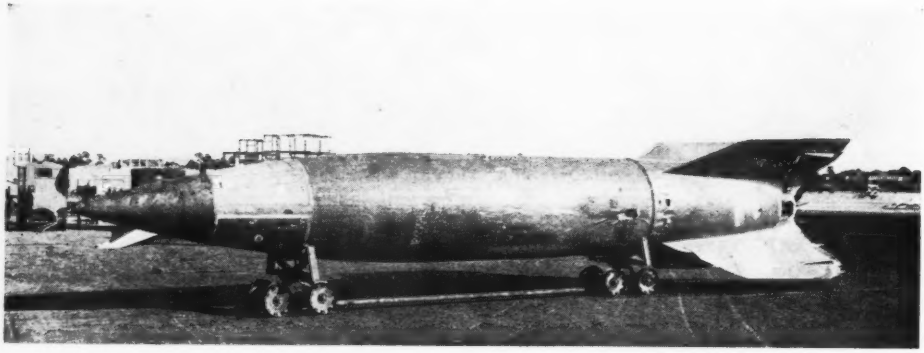


Fig. 1
The German A.4 Long-Range Rocket.

fuel rockets, employing liquid oxygen as the source of oxygen for the combustion of the fuel. Work was started on a range of rockets which they designated A1 to A10, only one of which, namely, A4 or as we know it V2, was ever used operationally.

Walter, working at Kiel, concentrated attention on H_2O_2, *i.e.*, hydrogen peroxide, called by the Germans T Stoff, and with this fuel in combination with a suitable catalyst, he developed forms of rocket which were used quite extensively by the Germans during the war.

Apart from these main rocket developments, there were many other developments going on all over Germany; in some of these solid fuels were used, and in others nitric acid was the oxygen carrier, and Diesel oil or alcohol the other fuel.

It would be impossible in the space of this lecture to touch on all these developments, indeed since they all contain the

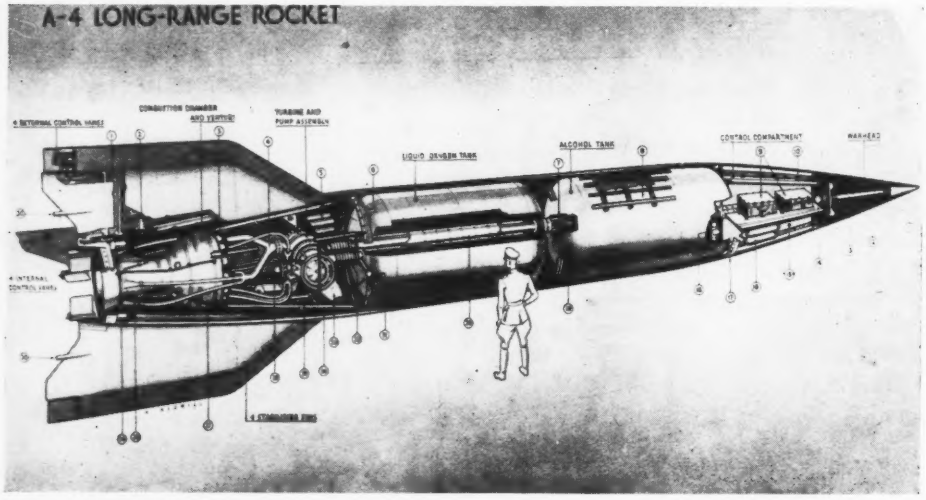


Fig. 2
Sectional View of Rocket.

same essential features, it will be sufficient to describe only one of them in detail. I propose therefore to describe the V2 rocket, and discuss the various factors affecting its performance, and only touch briefly on the other lines of development that went on in Germany.

V2 ROCKET

Early Work. The V2 rocket was the product of Prof. V. von Braun and his collaborators; Braun incidentally recognised that the V2 is still in a very undeveloped state, he is well aware of its shortcomings, and compares the stage of development reached with, say, the stage of development reached by the aeroplane in the last war.

Early work leading up to the V2 began in 1933, when the A1 rocket was designed. This rocket weighed about 330 lb., it was 4.6 feet long and had a diameter of about 1.0 feet. It was followed in 1934 by work on A2, which in dimensions was similar to A1, and differed only as regards the method of control. The A2 rocket fitted with a motor developing a thrust of 660 lb. for 16 seconds, was successfully launched, and reached a height of 6,500 feet. Developments on A3 started in 1938. A3 weighed 1,650 lb., was 25 feet long, and about 2.5 feet diameter. This rocket was fitted with a motor developing a thrust of 3,300 lb. for 45 seconds, it contained an automatic control system which operated rudders in the gas stream, and when fired vertically, it reached a height of nearly 40,000 feet, while it had a range of about 11 miles when launched at an angle.

Work on A4 commenced about 1940. A4 was a scaled up version of A3; firings commenced in July, 1942, and the first successful launch was in October, 1942, when the fourth rocket to be launched flew a distance of 170 miles. Orders for the production of the A4 rocket in quantity were given at the end of 1942, and the first attacks against this country started in September, 1944. Information was, however, in our

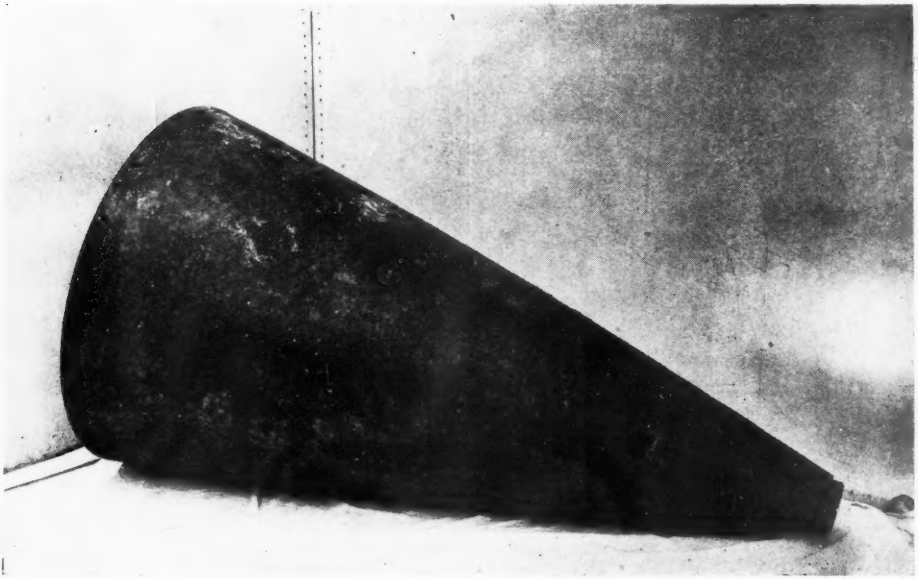


Fig. 3
Warhead.

Fig. 4
Radio and Instrument Bay.

hands well in advance of this date. Experimental rockets sometimes have a habit of departing from their proper course, and this happened in the case of an A4 rocket fired by the Germans in June of 1944. This particular rocket, as I think most of you now know, air burst over Sweden. In so doing, it yielded two tons of rather small fragments, but upon examination these revealed most of the design features and enabled us to establish the rocket size and performance.

The early experience with these experimental rockets is quite interesting, in that it illustrates the difficulties that have to be overcome in a development of this kind. The first rocket fired on 6th July, 1942, rose about three feet from the ground, and then exploded, destroying the testing station. The second rocket rose 16,000 feet into the air and then exploded, and the third one suffered a similar fate. The fourth, as I have just mentioned, flew satisfactorily, so also did the fifth, though this is uncertain because the point of impact was not spotted. Then followed a succession of failures; 13 rockets one after the other failed to fly, some broke in two, others exploded, and none of them reached its target. During the development which followed, well over a hundred rockets were fired, and although at the end of this period break ups were less frequent, failures still occurred to some 15-20 per cent. of the rockets fired.

GENERAL DESCRIPTION

The leading particulars of the A4 rocket are set out in the following table, and the general features can be seen from Figs. 1

and 2. The rocket has an overall length of 46 feet, a maximum body diameter of about 5 ft. 5 ins. and an initial all up weight of about 12.5 tons.

TABLE I.

LEADING PARTICULARS OF THE A4 ROCKET.

Lengths—	
Overall length of rocket	46 ft.
Warhead	5 ft. 11 ins.
Instrument bay	4 ft. 8 ins.
Fuel compartment ...	20 ft. 3.5 ins.
Power bay (including Venturi)	14 ft. 7.5 ins.
Diameters—	
Maximum diameter of body	5 ft. 5.3 ins.
Diameter over fins ...	11 ft. 8 ins.
Weights (for detail weight summary see Appendix I)—	
Total loaded weight ...	12.5 tons.
Warhead (with Amatol filling)	2,150 lb.
Structure	3,865 lb.
Power unit	2,235 lb.
Equipment	650 lb.
Main fuels (tanks full)...	19,310 lb.
Auxiliary fuels	400 lb.

The Warhead, Fig. 3, occupies the nose of the rocket, and housed immediately behind the Warhead are the main control instruments and radio equipment. The outer shell of the rocket extending over this compartment, Fig. 4, is provided with large hinged panels, affording access to, and a ready means of servicing, the instruments carried. The main fuels are carried in two large light alloy tanks, Fig. 5, occupying the central compartment of the rocket. There are two fuels, one a 75 per cent. solution of ethyl alcohol in water, occupying the forward tank nearest the nose, and the other fuel, liquid oxygen, occupying the rear tank.

The remaining space in the rocket is taken up by a turbine which drives the two main fuel pumps, and with the auxiliary fuel supply, then follows the main combustion chamber and exit venturi; around this unit are the stabilising fins, while projecting into the exit of the venturi are the main control vanes, Figs. 6 and 7.

Structure.—The structure of the rocket, Fig. 8, is built up of steel ribs and stringers which are covered by thin steel sheeting. The sheeting is in general 0.025 inches thick,

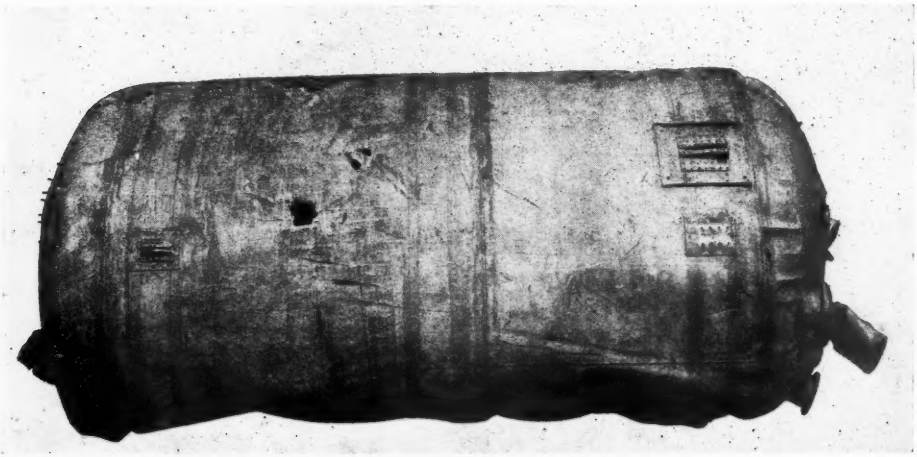


Fig. 5
Alcohol Fuel Tank.

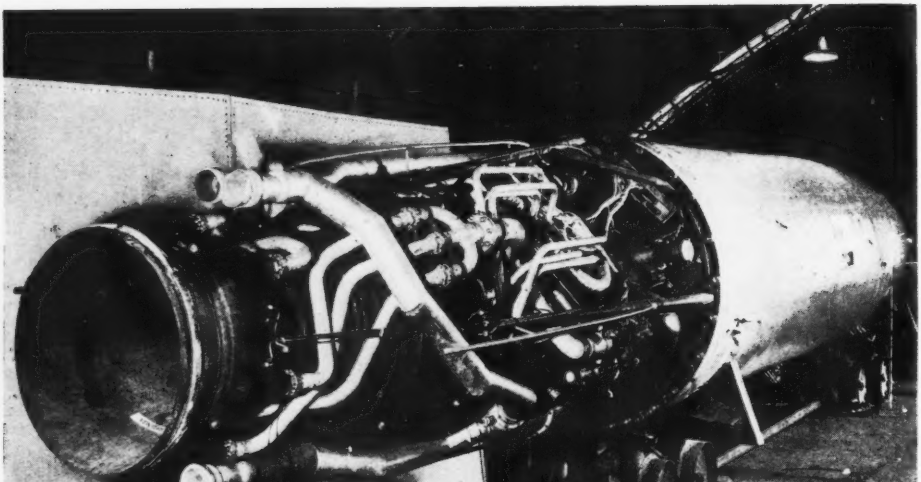


Fig. 6
Motor Compartment and Venturi.

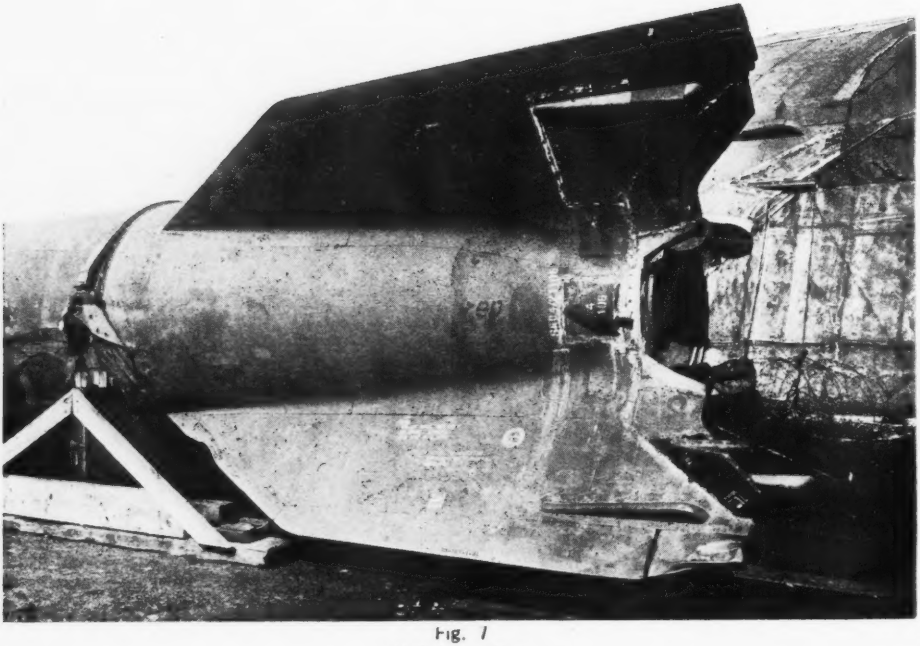


Fig. 7
Stabilising Fins

but the stringers and formers vary, and their thicknesses range from 0.04 to 0.068 inches. The constructional features, as will be seen from the figure, resemble closely normal aircraft practice.

The thrust of the jet is transmitted from the venturi unit, through a heavy tubular steel framework, to a circular angle frame at the forward end of the power compartment. This framework also forms the main supporting structure for the turbine and pump units.

The main fuel tanks are positioned by guides built into the outer skin covering, but the loads are taken directly by the circular angle frames, which form the ends of the fuel compartment. Provision is made to take up the very considerable differential expansion between the tank and outside shell cover that will occur during the filling process, and which will become even more accentuated during the rocket flight.

Forward of the tank compartment the structure changes, and over the space occupied by the control instruments and radio the rocket tapers, and the strength is provided by four main longitudinal members, which again terminate in a circular angle frame.

A conically shaped Warhead forms the nose of the rocket, and this is bolted to the circular frame at the end of the instrument bay. The Warhead shell is of steel, about 0.25 inches thick, and it is not protected in any way from the conditions that will be experienced by the outer skin of the rocket during flight. The main circular angle frames, mentioned above, act as transport joints, thus enabling the rocket to be broken

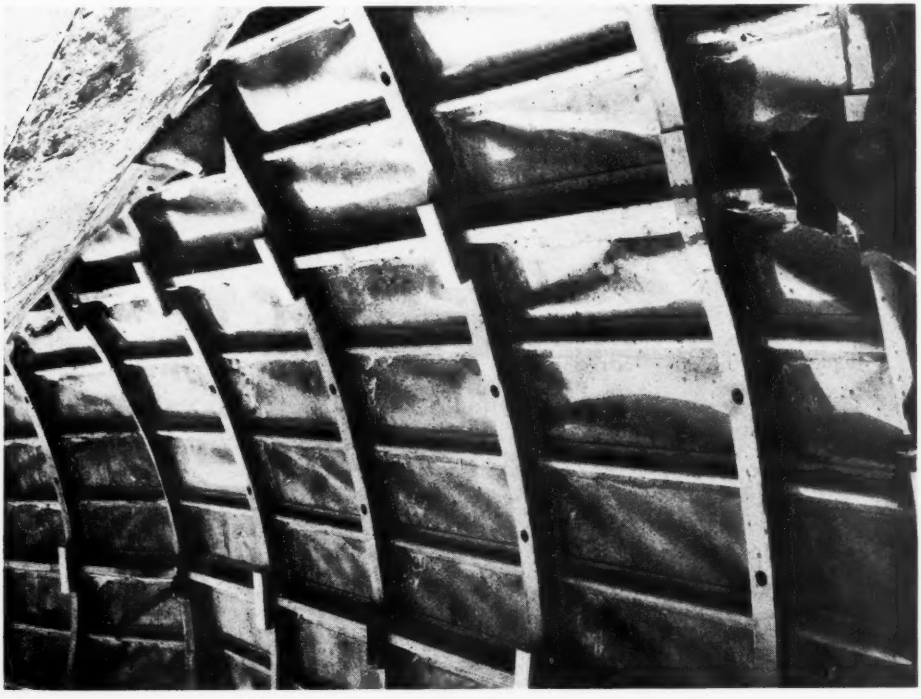


Fig. 8
Structure in neighbourhood of Fuel Tanks.

down into its various components for ease of handling.

Propulsive Unit and Fuel System

The rocket is propelled by a jet of hot gases resulting from the combustion of two fuels, liquid oxygen and alcohol, the combustion taking place in a chamber which forms the front end of the venturi.

The fuels are pumped under pressure into this space by two turbine-driven centrifugal pumps, the turbine being driven by a hydrogen peroxide-permanganate system.

Fig. 9 shows the layout of the combustion chamber and pumping system, and Fig. 10 the arrangement of the auxiliary system. Hydrogen peroxide contained in an elliptical steel pressure tank, and sodium permanganate contained in a small cylindrical tank are fed by means of compressed nitrogen to a burner. The hot gases generated are then led along a lagged steel pipe to the ring main which feeds the turbine.

The turbine, Fig. 11, has one velocity compound impulse stage, with the rotor carrying two rows of blades. The turbine has partial admission to both rows of blades,

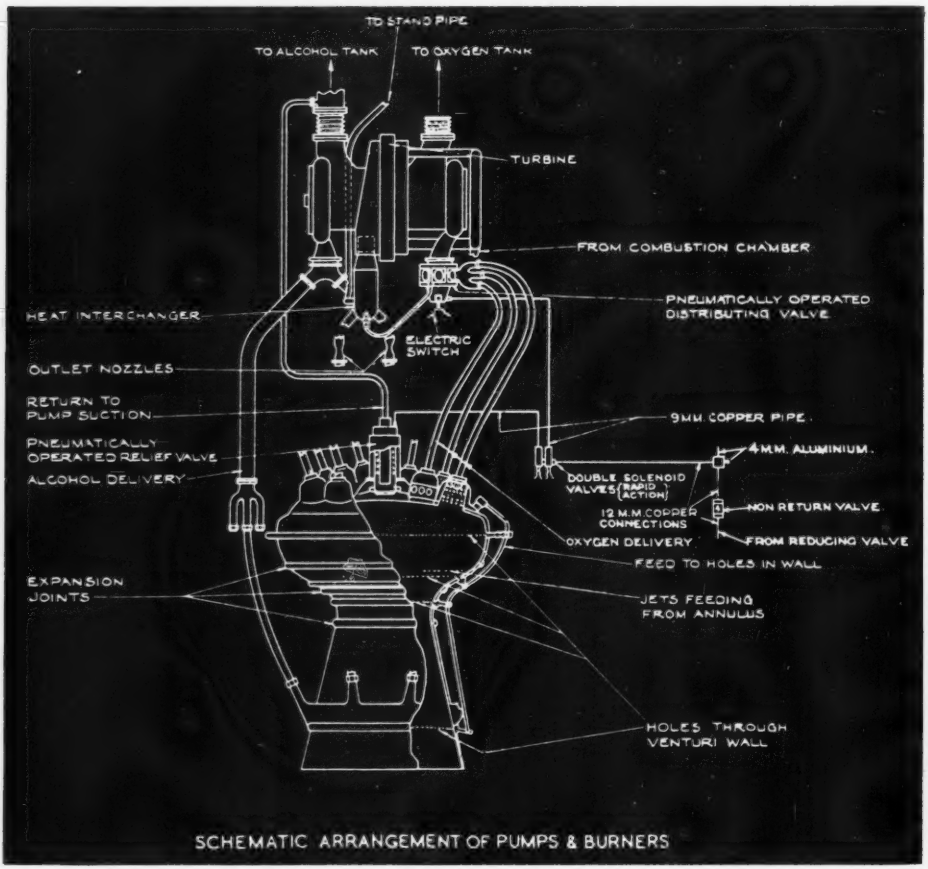


Fig. 9

the inlet nozzles to the first blades being arranged in four segments, each with four nozzles per segment.

Each nozzle is of the convergent-divergent type, and the ratio of the throat to outlet area, corresponds to a pressure ratio of about 20 to 1. The turbine drives two centrifugal pumps, Figs. 12 and 13, and the performance of the turbine and pumps under operating conditions are set out in Appendix II. These figures are estimates, but they agree fairly closely with statements on performance that have been put forward by various German workers.

The pumps deliver about 275 lb. of fuel per second at about 350 lb./in.^{2} pressure, and at this rate of discharge the main tanks would be exhausted in about 70 seconds.

Both main tanks are pressurised to about 1.4 atmospheres, partly to assist the pumps, and partly to prevent the tanks collapsing. Pressure in the alcohol tank during the initial stage is achieved by ram compression, a pipe from the tank being lead forward through the Warhead to an opening near the nose. After about 40 seconds, however, owing to the fall off in air density this method of pressurising is no longer effective; a valve in the pipe is then closed, and pressure is provided by supplying nitrogen

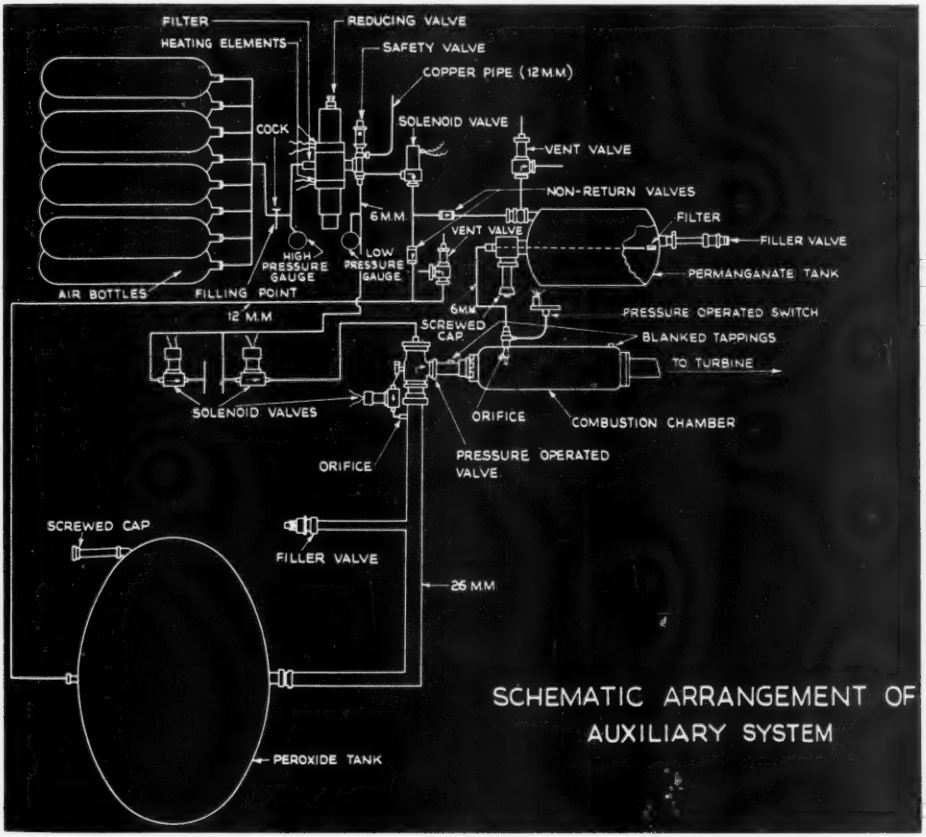


Fig. 10

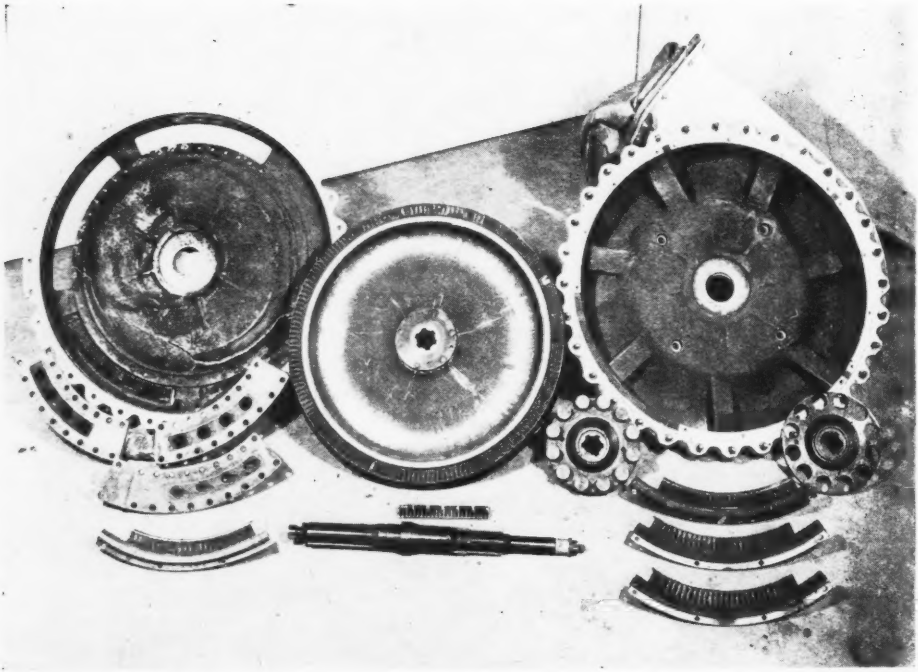


Fig. 11
Turbine—Dissembled.

from a group of bottles housed in the instrument bay.

On the oxygen tank a special vent valve is fitted that maintains the pressure in the tank at 1.2 atmospheres, but during the operational period this pressure is raised to 1.5 atmospheres, the pressure being kept up by supplying oxygen which is evaporated by a heat exchanger placed in the exhaust of the turbine system.

Fuel from the pumps is passed to distributing valves, the liquid oxygen being fed to eighteen brass roses, located at the base of open-ended cups formed on the end face of the combustion chamber, while the alcohol is pumped to an annular ring at the rear end of the venturi, where it enters a cooling jacket which surrounds the venturi. The venturi, Figs. 14,* 15 and 16, is a propulsive nozzle of the convergent divergent type. It is of heavy welded mild steel construction, a large portion of it being jacketted, to accommodate the alcohol coolant. Most of the cooling is external, and the alcohol after passing between the double walls is finally lead to the eighteen cups on the end face of the combustion chamber, where it discharges through a ring of small nozzles and mixes with the liquid oxygen.

A small amount of alcohol, however is fed into the venturi through rings of small holes drilled at several positions along its length, thus providing some degree of internal wall cooling. Although the gas tem-

* Fig. 14, which is a photograph of the venturi of the Swedish rocket, shows a venturi which is rather different to the production type; the main face of the combustion chamber being formed in light alloy and bolted to the remaining steel part of the chamber.

peratures in the combustion chamber and near the throat of the venturi reach 2,700-2,500°C., the cooling is very effective, since examination has shown that the skin temperature on the inner surface has not exceeded 1,000°C. The only trouble that seems to have arisen is a thermal expansion one, and small loops have been introduced into some of the feed systems to overcome this difficulty.

Rocket Motor Characteristics

Before going on to discuss the controls and control systems, it will perhaps be best to say something about the performance of the rocket motor and the way it was fired.

If we consider a simple system comprising a combustion chamber and venturi, Fig. 17, in which the fuels are burnt at constant pressure, and if we assume as the result of the combustion process, that the gases have

Fig. 12
Turbine and Pumps.

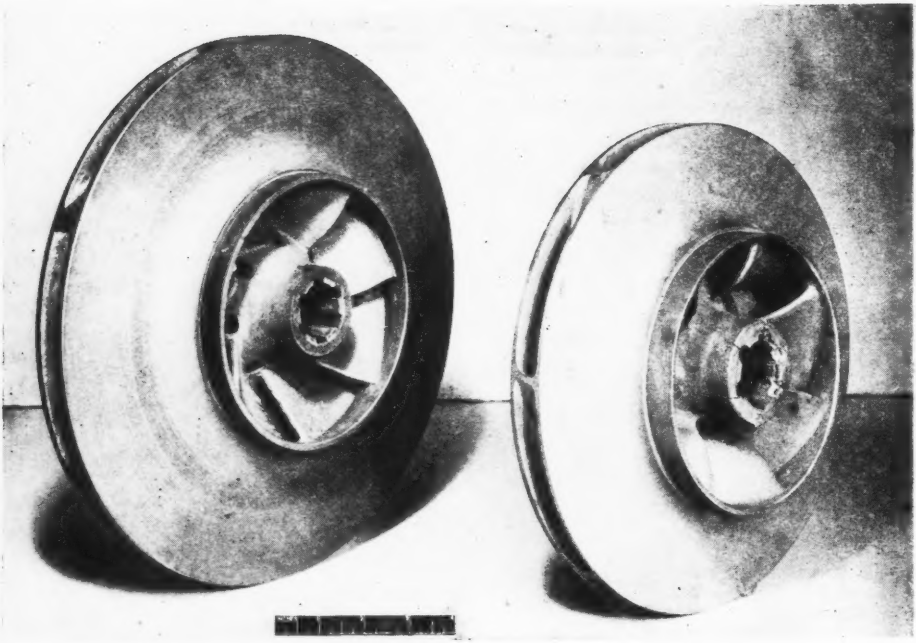


Fig. 13
Pump Rotors.

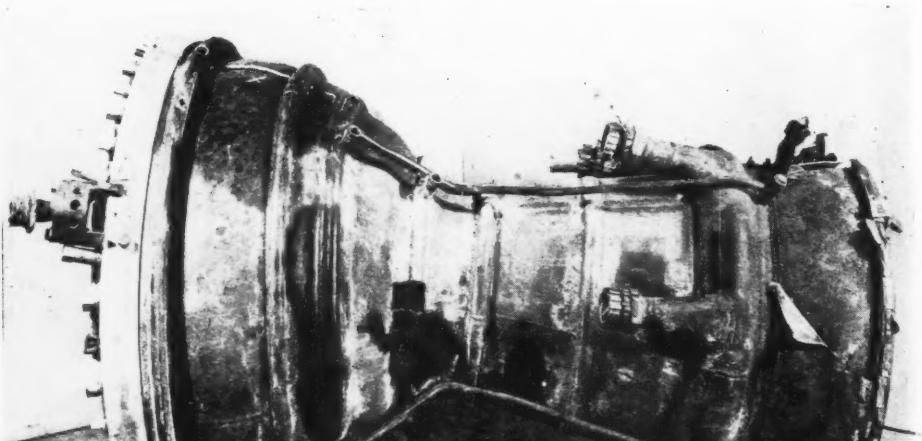


Fig. 14
Venturi.

a pressure, density and temperature given by p_o, ρ_o and T_o, then by making use of the continuity relationship

$$S\rho v = M$$

where S is the cross sectional area at any point, v is the velocity and M is the rate of discharge, and of the Bernoulli relation

$$v^2 = \frac{2\gamma}{\gamma-1}\frac{p_o}{\rho_o}\left(1-\frac{p}{p_o}\right)^{\frac{\gamma-1}{\gamma}}$$

together with the adiabatic law and gas laws which give

$$\frac{p}{p_o} = \left(\frac{\rho}{\rho_o}\right)^{\gamma} = \left(\frac{T}{T_o}\right)^{\frac{\gamma-1}{\gamma}}$$

it is possible to establish a relation between the shape of the venturi and its operating conditions, one form of which can be written

$$M = S\left(\frac{2}{\gamma-1}\right)^{\frac{1}{2}} a_o\rho_o\left[\left(\frac{p}{p_o}\right)^{2}_{\gamma} - \left(\frac{p}{p_o}\right)^{\frac{\gamma+1}{\gamma}}\right]^{\frac{1}{2}}$$

where γ is the ratio of the specific heats and

$$a_o^2 = \gamma\frac{p_o}{\rho_o}$$

is the speed of sound in a gas corresponding to the pressure and density $p_o\rho_o$. Fig. 18 shows some of these relationships for an ideal venturi, when $\gamma = 1.25$.

In the case of the A4 rocket, the ratio of the discharge area to the throat area is 3.4 to 1, so that $\sqrt{S_x/S_1}$ is 1.85 and consequently the ratio of the pressure in the combustion chamber to the pressure at exit must be about 20 to 1 (see curve 1 of Fig. 18), if the venturi is to run full.

The conditions in the combustion chamber are of course dependent on the rate of delivery of the fuel, and on the characteristics of the fuel, and for the rocket to operate under ground level conditions with the gases discharging at atmospheric pressure, sufficient fuel must be burnt to maintain the pressure in the combustion chamber at 20 atmospheres, *i.e.*, 294 lb./in.^{2}, and the fuel pumps must be capable of maintaining the fuel flow at this pressure in addition to providing the pressure head necessary to overcome the losses in the fuel lines from the pumps and the losses through the nozzles. When the combustion chamber is subject to a pressure p_o, it is clear there is a nett unbalanced reaction equal to S_1p_o propelling the rocket forward, since however the venturi converts some of the internal energy of the gases into useful work, the actual thrust will be greater than this value.

In discussing rocket fuels, it is often convenient to refer to their specific impulse (S_1), *i.e.*, the thrust produced for every pound of fuel burnt per second. The thrust therefore can be written

$$T = M\,(S_I)$$

but the thrust can also be written

$$T = M\frac{v_x}{g}$$

where v_x is the discharge velocity, it follows therefore that

$$(S_I) = \frac{v_x}{g}$$

From the velocity relationship given earlier, we may write

$$v_x = \sqrt{\frac{2\gamma}{\gamma-1}\frac{p_o}{\rho_o}\left(1-\frac{p_x}{p_o}\right)^{\frac{\gamma-1}{\gamma}}}$$

and this reduces to

$$v_x = \sqrt{\frac{2\gamma}{\gamma-1}\frac{p_o}{\rho_o}}$$

since the part in brackets tends to unity.

Consequently the specific impulse of a fuel can be written

$$S_I = \frac{1}{g}\sqrt{\frac{2\gamma}{\gamma-1}\frac{p_o}{\rho_o}}$$

or making use of the gas law

$$\frac{p}{\rho} = \frac{RT}{M}$$

where R is the gas constant, and M the molecular weight

$$S_I = \frac{1}{g}\sqrt{\frac{2\gamma}{\gamma-1}\frac{RT_o}{M}}$$

The specific impulse, therefore, varies as the square root of the absolute temperature of the gases in the combustion chamber, and inversely as the square root of the molecular

weight. The pressure, however, in the combustion chamber varies roughly as the fifth power of the temperature, so that there are very severe practical limits to the gain that can result from higher working temperatures; but it is an advantage to work with gases having low molecular weight.

The fuel used in the A4 rocket has an (S_1) of about 220 under ground operating conditions; consequently the velocity of discharge of the gases is about 7,000 ft./sec., and since 275 lb. of fuel are burnt every second, the thrust developed by the rocket will be about 60,000 lb. The approximate

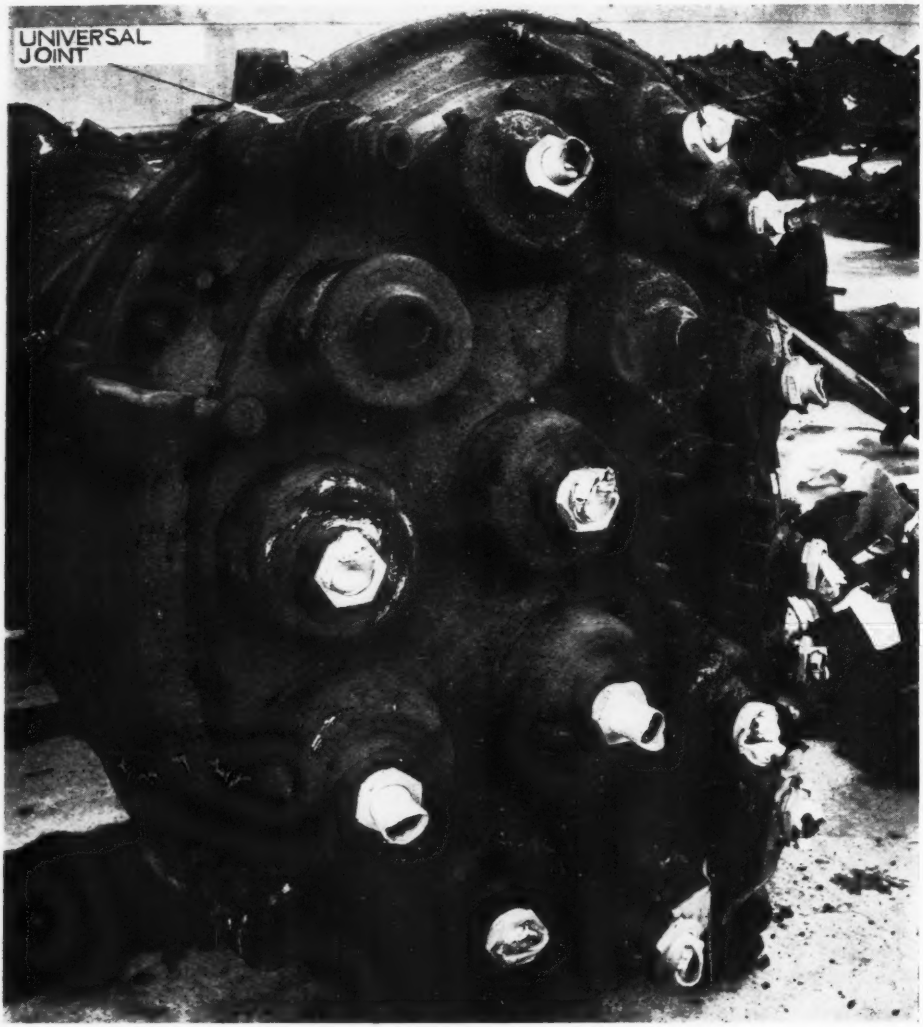


Fig. 15
Burner Cups

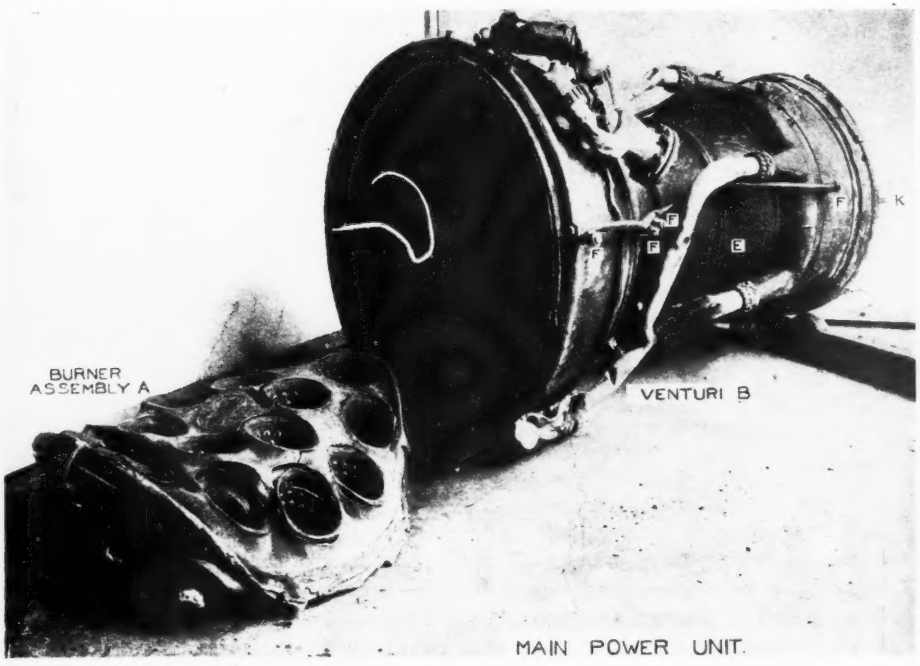


Fig. 16
Burner Assembly and Venturi.

working conditions within the venturi are as indicated in Fig. 17.

The firing of the rocket is carried out with the rocket in the vertical position. The rocket is placed in the vertical position while it is still empty and it is then filled with fuel. Before the actual filling, each tank is pressurised to 1.8 atmospheres for five minutes to check for tightness, after which the fuels are pumped into the rocket from mobile wagons.

The fuels are pumped in the following order:—

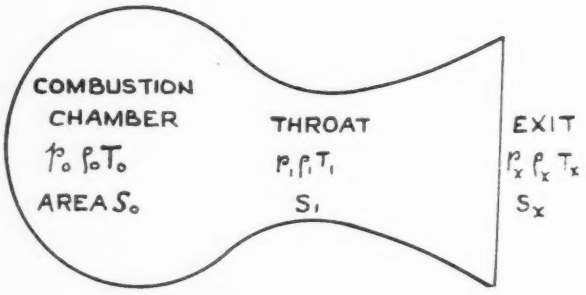
Alcohol,
Hydrogen peroxide,
Liquid oxygen,
Permanganate.

The filling operation takes only 12 minutes, and it is one of the last operations before firing, as about 4.5 lb. of oxygen per minute are lost by evaporation, if the rocket is filled and left standing.

Two forms of igniting torch were tried, one a pyrotechnic torch (black powder) and the other a liquid operated torch, the former being preferred because it required less attention.

When the firing torch is alight, the main alcohol and oxygen valves are opened. These and the other valves throughout the rocket are operated by compressed nitrogen supplied from a bank of bottles situated in the power compartment, the same supply being also used to pressurise the auxiliary fuel tanks and force these fluids into their mixing chamber.

With the main fuel valves open 20 to 30 lb. of alcohol and oxygen per second are fed by gravity through the pumps to the combustion chamber, and are then ignited,



APPROXIMATE CONDITIONS IN VENTURI

	Combustion Chamber	Throat	Exit
Diameter (ft.)	3.11	1.32	2.41
Pressure (lb./in.² ABS) ...	294	157	14.7
Temperature (°K.)	3,000	2,670	1,650
Velocity (ft./sec.)	0	3,400	7,000

Fig. 17
Conditions in Combustion Chamber and Venturi.

the burning taking place without shock. Combustion is maintained for a few seconds, until an observer stationed at a distance is assured conditions are satisfactory; he then makes an electric contact which starts up the flow of auxiliary fuel, the turbine is started and is stated to reach full speed in about 3 seconds. During this time the flow of the main fuels to the combustion chamber increases, the thrust builds up, and soon exceeds the weight, causing the rocket to commence its vertical ascent. The whole operation, from the instant of firing the torch to the rocket unit being up to full thrust, taking only from 7 to 10 seconds.

Control

For maximum range it is necessary to turn the rocket on to its course and rotate it from the vertical until it takes up an angle of about 40 to 45° to the horizontal, by the time it has reached the all-burnt stage, *i.e.*, after 60 seconds flight. To do this the rocket is fitted with four graphite controllers, Fig. 19, that are symmetrically placed around the exit of the jet, and four external controllers that are carried at the tips of the stabilising fins, Fig. 20.

The internal controllers are each operated by separate electric-hydraulic servo units. The two internal and external controllers which lie in the plane of rotation of the turbine rotor, being connected through a claw fitting; each internal controller and its corresponding external controller operating in the same sense. Electrical coupling is provided between the two servos driving these controllers, such that the controllers can be employed to provide roll and azimuth control.

The other controllers, *i.e.*, those lying in a plane at right angles to the plane of the turbine rotor, have a separate drive each; these drives are, however, electrically synchronised so that all the controllers move up or down together, and control the movement in the pitch plane.

The form of the control unit has undergone a good deal of change since the first rockets were fired against this country, but broadly all systems attempt:—

(i) Having set up the rocket with the plane of the turbine rotor lying in the plane of the target, to maintain the rocket axis in this plane.

(ii) Stabilise the rocket in roll.

(iii) Rotate the rocket axis in pitch at some predetermined rate.

(iv) Obtain some measure of the rocket velocity, and so enable the main fuel to be cut off at some point dependent upon the desired range.

When setting up the rocket for firing it was placed on a small horizontal turntable and the table was rotated so as to bring the pitch axis of the rocket perpendicular to the plane containing the target. This setting up was done by surveying methods. To control the motion of the rocket, in azimuth, roll and pitch, an automatic pilot was fitted. One form of this pilot, Fig. 21, comprised two electrically driven gyros, one with its axis along the axis of the rocket, and the other with its axis perpendicular to the axis of the rocket, and to the plane containing the target. This second gyro detects the roll and yaw of the rocket, but is unaffected by the pitching motion: the roll and yaw is measured by means of two fine wire potentiometers, and their outputs after passing through an electronic amplifier operate the controllers and so restore the rocket to its correct course.

The other gyro provides a pitch control, and during flight this gyro is made to precess towards the horizontal in the roll and yaw plane and, as in the case of the roll and yaw gyro, the movement is detected by means of a fine wire potentiometer, the output from the potentiometer is fed into an amplifier, and is then made to operate the controllers through their appropriate servos. In the form shown in Fig. 21, the pitch control unit consists of five independently adjustable resistances in series with the pitch descriminator coil. Four of these resistances are initially shorted, and as they are brought in in succession, they give five different rates of precession to the gyro.

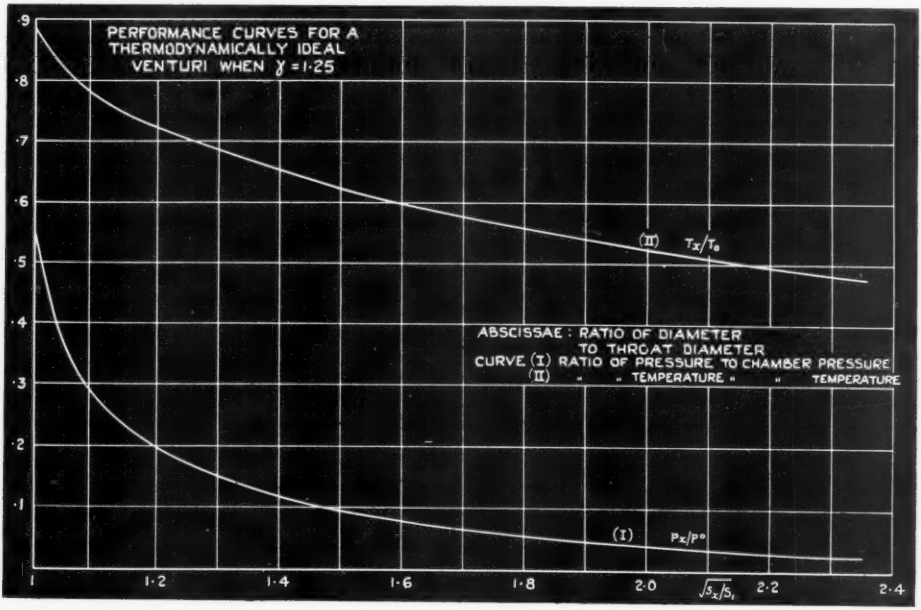


Fig. 18
Performance Characteristics of Venturi.

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Fig. 19
Graphite Controller.

In some of the early rockets this function was carried out by the rotation of a drum switch driven by a constant speed motor and contained within the pitch gyro case, but in later versions the leads are lead out of the case to an external switching device.

One unit that has been examined was set to precess the gyro, and so turn the rocket in the following way:—

Time from start in seconds.	Angle.
0	90°
2.0	90°
8.5	80°
11.5	76°
28.5	60°
47.0	46°
52.0	44°

the rocket would therefore be turned from the vertical through 46° in 52 seconds.

The amplifier fitted into the rockets provided for differentiation of the input signal and so introduced a rate term into the control; it also provided for the mixing of the roll and yaw signals so that both functions could be controlled by the one set of vanes; a feedback from the pitch control vanes also ensured their synchronisation.

The automatic pilot, just described, meets three of the requirements mentioned above, leaving the control of range.

In examining the question of range, it can be shown that in the case of a rocket travelling with its axis at about 45° to the horizontal, the range is dependent primarily upon the height and speed, and is not very sensitive to small variations of the actual angle. Furthermore, if the thrust characteristics of the rocket motor are reasonably repeatable, the height and velocity during the initial part of the flight will be related, since the pitch gyro will control the rocket path against time. It follows, therefore, that if the velocity can be measured, control of velocity can be used to control range.

This has been the basis of the method employed and early rockets contained two radio sets, one a receiver transmitter, and the other a receiver for velocity measurement and range control. The velocity was measured by a doppler effect, a signal sent out from the ground was received and retransmitted by the rocket at twice the received frequency. By doubling the outgoing signal and beating it with the returned signal, a beat frequency was obtained which is a direct measure of the rocket velocity. When the rocket reached a velocity which was dependent on the range desired, a signal was sent to the second receiver in the rocket, and this caused the turbine to be shut down, the fuel supply was therefore cut off and the rocket proceeded without further control, rather like a shell fired from a gun.

Elaborate security arrangements were built into the radio circuits, so as to prevent the rocket motion being influenced by false signals.

The radio method of measuring velocity was soon dropped in favour of an integrating

accelerometer, Fig. 22, which was carried in the rocket, and so made the rocket self-contained and independent of any ground control from the moment of taking off on its flight.

The integrating accelerometer consists of a pendulous gyro mounted in gymbals and operating two sets of contacts whose angular separation can be varied. The gyro case carries a centralising switch, the leads of which run to a reversible induction motor geared to the gymbal. Any deviation from the mid-position of the gyro axis causes a torque to be applied to the gymbal in the sense appropriate to precess it back again. Because the centre of gravity of the gyro does not lie on the axis *CD*, any acceleration of the system parallel to the axis *EF*, *i.e.*, the rocket axis, will cause precession about this axis at a rate proportional to the acceleration. The angle of precession will be proportional to the integral with respect to time of all accelerations, including any component of the earth's gravitational field acting in the direction *EF*.

Rotation of the outer gymbal, which actuates the contacts, therefore represents the value of the quantity

$$\int\left(\frac{dV}{dt}+g\sin\theta\right)dt$$

$$\text{i.e., } V+g\int\sin\theta dt$$

and so affords a measure of the rocket velocity; θ being the angle of the rocket axis to the horizontal.

The contacts can be made to control one function in the ground installation at starting, and two functions, slightly separated in time, when the quantity above has reached a preset value.

These functions are thought to be opening up to full thrust at take-off, and throttling first to about 1/3 thrust, and finally to zero thrust when the required velocity has been reached.

The shutting down from full to 1/3 thrust took about two seconds and was done to avoid difficulties due to " water hammer," the final shutting down to zero thrust took a further two seconds, and these times had of course to be taken into account in setting the contacts on the integrating accelerometer.

Characteristics of Trajectory.

The typical rocket trajectory is shown in Fig. 23. The early path, during which the motion is controlled by the pitch gyro is very nearly an arc of a circle, the rocket reaching a height of 22-23 miles and a velocity of about 5,000 ft./sec. (3,400 m.p.h.) after about 60 seconds flight, as it is nearing the " all-burnt " condition.

It should be noted that during most of this period it is the internal controllers that provide the main control. At the start of the flight there is no forward velocity over the external controllers, so they cannot exercise any control, while near the all-burnt stage, although the rocket is moving at about 5,000 ft./sec., the relative density has fallen to 0.0053, and the indicated air speed is only 360 ft./sec., so that again the control provided by the external controllers is small. At some intermediate height when they are effective, the inner controllers are equally effective, and can provide all the control that is necessary so that it is a little difficult to understand why the Germans went to the complication of fitting these external controllers.

After all-burnt the rocket path is parabolic, and the velocity falls as the height increases, the maximum height reached being about 60 miles. How small the forces on the rocket are at this height can be judged from the fact that near the top of its trajectory the indicated air speed falls to about 3.5 ft./sec., although the rocket is still travelling at nearly four times the local speed of sound. Once over the top the speed again

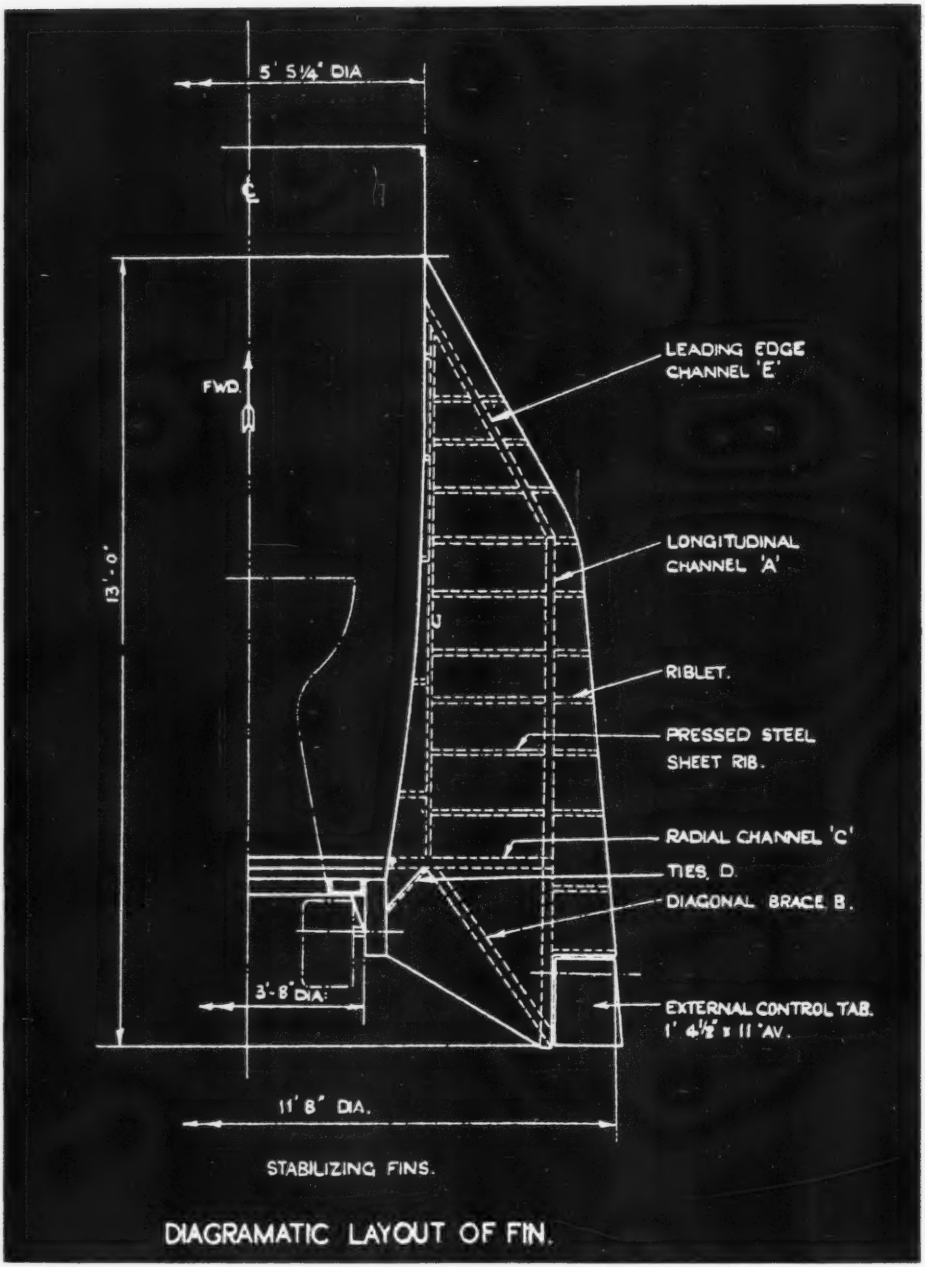


Fig. 20
Stabilising Fin and Controllers.

rises and reaches a second maximum as the rocket moves into the denser air.

The actual operational range of rocket varied widely, a few rockets travelled as far as 220 miles, but most of them averaged 180-190 miles. Since the main path is parabolic, the height reached will always be about quarter of the range.

The total time of flight is about five minutes. The highest acceleration reached, which is at all-burnt when thrust has risen to 69,000 lb. and the weight has fallen to 8,900 lb., is just under 8g.

The highest velocity reached is at all-burnt, the rocket is then travelling at 5,000 ft./sec. and it is interesting to note that at this velocity the rocket motor is developing well over 600,000 horse-power. The stagnation temperature corresponding to a velocity of 5,000 ft./sec. is about 1,400°K, and in view of this it might be expected that the skin temperatures of the rocket during flight would tend to be high and approach the stagnation value.

To check this point a careful examination of the rocket skin was undertaken, and both

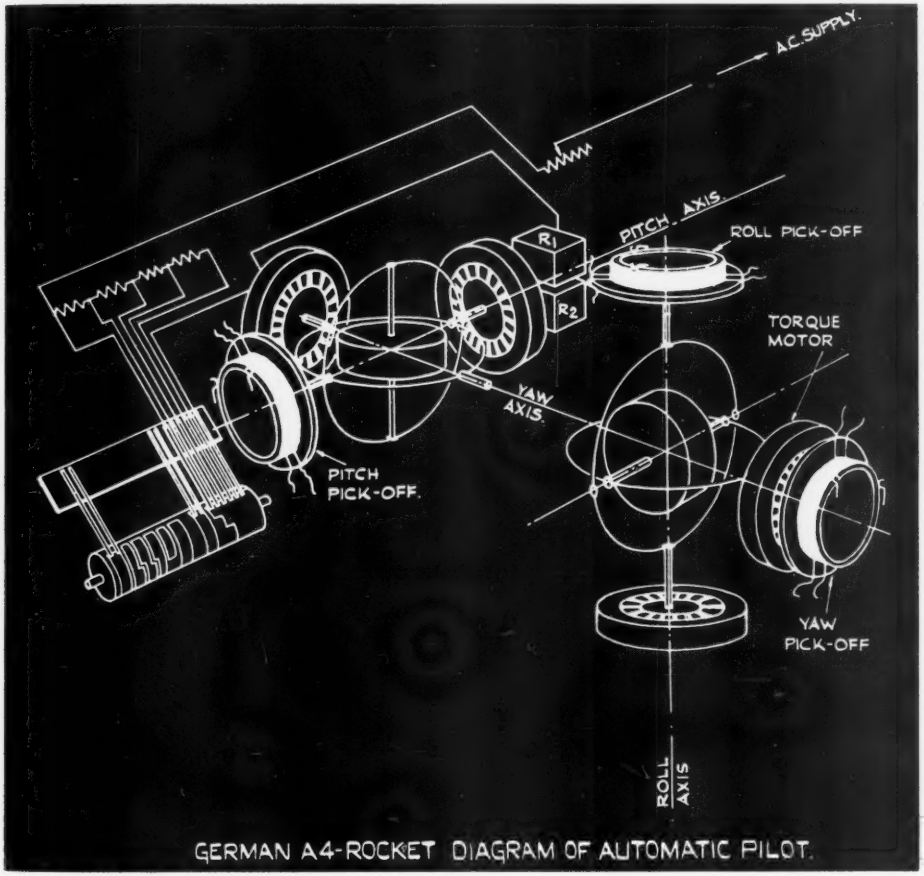


Fig. 21

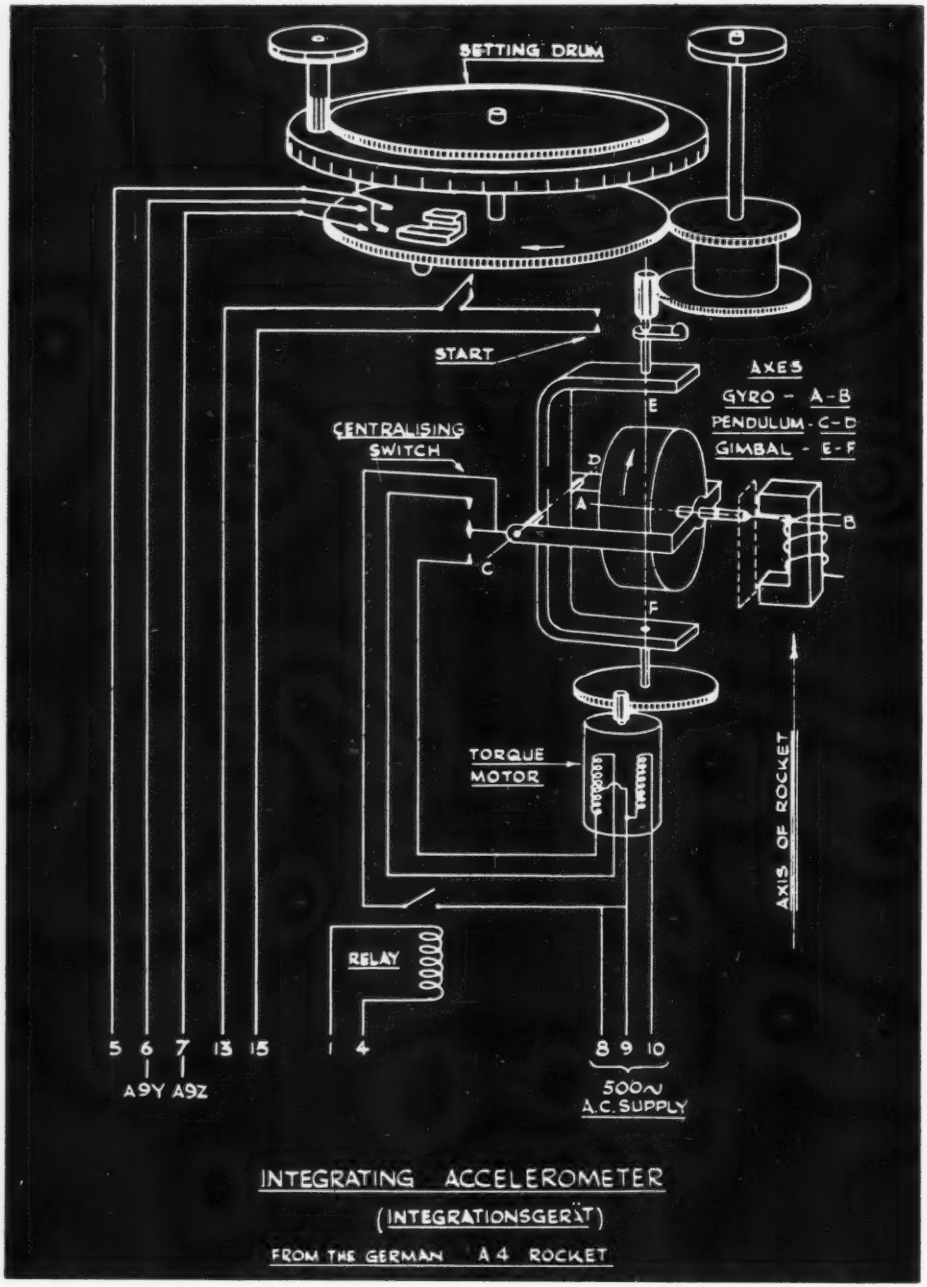

SETTING DRUM
START
AXES
GYRO - A-B
PENDULUM - C-D
GIMBAL - E-F
CENTRALISING SWITCH
D
A
C
E
F
B
TORQUE MOTOR
AXIS OF ROCKET
RELAY
5 6 7 13 15
A9Y A9Z
1 4
8 9 10
500~ A.C. SUPPLY
INTEGRATING ACCELEROMETER
(INTEGRATIONSGERÄT)
FROM THE GERMAN A 4 ROCKET

Fig. 22

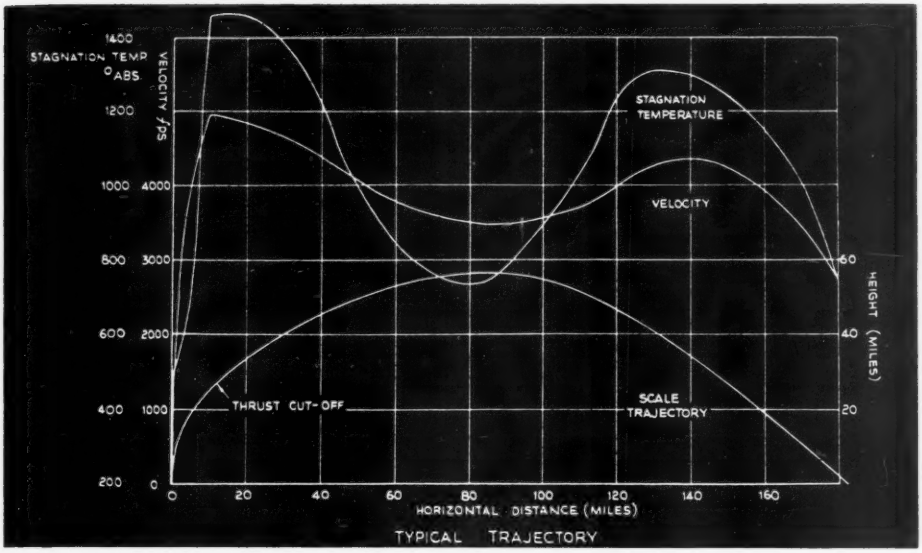


Fig. 23

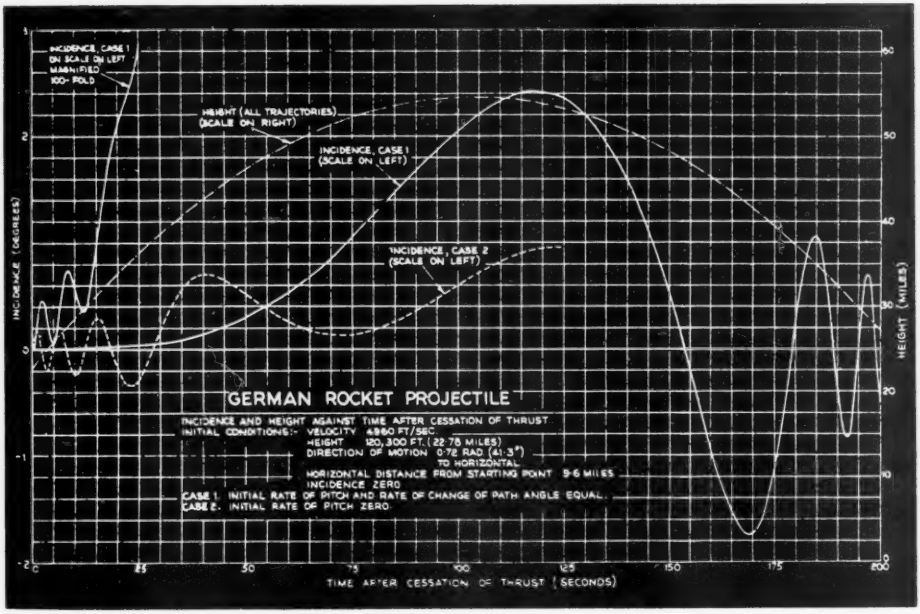
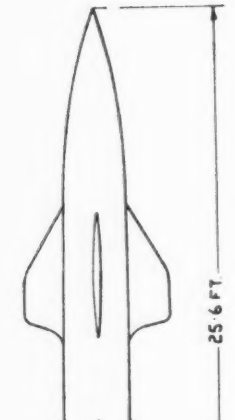
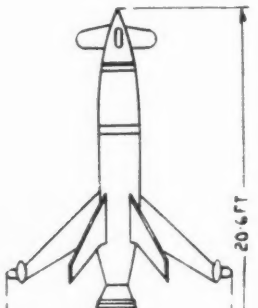

INCIDENCE, CASE 1 (SCALE ON LEFT) MAGNIFIED 100-FOLD
HEIGHT (ALL TRAJECTORIES) (SCALE ON RIGHT)
INCIDENCE, CASE 1 (SCALE ON LEFT)
INCIDENCE, CASE 2 (SCALE ON LEFT)
INCIDENCE (DEGREES)
HEIGHT (MILES)
GERMAN ROCKET PROJECTILE
INCIDENCE AND HEIGHT AGAINST TIME AFTER CESSATION OF THRUST
INITIAL CONDITIONS:- VELOCITY 4980 FT/SEC
HEIGHT 120,300 FT. (22·78 MILES)
DIRECTION OF MOTION 0·72 RAD. (41·3°) TO HORIZONTAL
HORIZONTAL DISTANCE FROM STARTING POINT 9·6 MILES
INCIDENCE ZERO
CASE 1. INITIAL RATE OF PITCH AND RATE OF CHANGE OF PATH ANGLE EQUAL.
CASE 2. INITIAL RATE OF PITCH ZERO
TIME AFTER CESSATION OF THRUST (SECONDS)

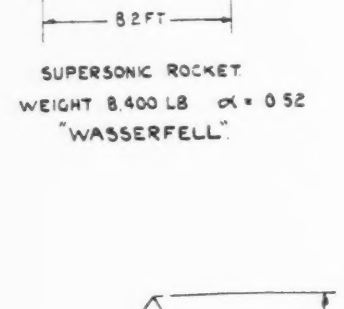
Fig. 24
Rocket Behaviour in Upper Atmosphere.



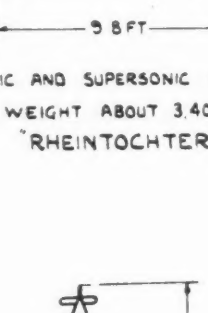
SUPERSONIC ROCKET
WEIGHT 8,400 LB α = 0·52
"WASSERFELL"



SONIC AND SUPERSONIC ROCKET
WEIGHT ABOUT 3,400 LB
"RHEINTOCHTER"



SUBSONIC ROCKET
WEIGHT 4,350 LB α = 0·38
"ENZIAN."



"SCHMETTERLING"
WEIGHT 970 LB. (WITH BOOSTER)
550 LB (BOOSTER JETTISONED)
α = 0·47

Fig. 25
German Anti-Aircraft Rockets.

from an examination of the condition of the paint, and a metallurgical examination of the skin material, it was concluded that the skin temperatures had not exceeded about 900°K. This figure agrees very closely with measurements that the Germans were able to make on an actual rocket in flight. In their tests the Germans inserted small discs of various metals of known melting point into the skin of the rocket and connected these into electrical circuits. As each disc melted, a signal was transmitted by telemetering to a ground station. It was found that the skin temperatures nowhere exceed 920°K, conduction and radiation losses therefore must have kept the skin temperature down well below the stagnation temperature.

In the course of our investigation into the A4, calculations were made to determine the stability of the rocket during its passage through the upper atmosphere. The results of these calculations are shown in Fig. 24. Two cases were considered, one in which the initial rate of pitch and rate of change of pitch angle were equal, and the other in which the initial rate of pitch was zero. The calculations were confined to a consideration of the behaviour of the rocket after all-burnt, and assumed that the rocket velocity was then 4,960 ft./sec. the height 22.8 miles, and the rocket axis made an angle of 41.3° to the horizontal.

It will be seen from the figure that under both assumptions, the rocket developed a pitch oscillation, which is undamped on the upwards parts of the trajectory, but which is quite heavily damped as the rocket descends into the denser atmosphere. The motion is never very large, and in neither case does the rocket incidence exceed 2.5° to the line of flight.

OTHER ROCKET DEVELOPMENTS

There were in addition to the A4 rocket just described a large number of other rocket developments going on in Germany. The A9 for example was an A4, equipped with wings, permitting it to glide when it reached the stratosphere, with a view to increasing the range.

A10 was a still more ambitious rocket, in the project stage only; it was to weigh about 85 tons, and was intended to carry the A9 rocket into the stratosphere, and then be jettisoned, A9 and A10, therefore, were separate stages of a two-stage rocket.

Towards the end of the war, when Germany was in serious difficulties because of the Allies very heavy bombing attacks, a great deal of attention was given to defence against these attacks and work on a large number of ground-to-air rocket weapons was started.

The number of these schemes started up seems to be closely related to the number of people with "bright ideas," and "Enzian," "Wasserfall," "Taifun," "Rheintochter," "Schmetterling" are some examples, Fig. 25.

Two other rocket applications, also intended as a defence against bombers, are the Me.163, and "Natter." These are interesting because both were piloted, though in the case of "Natter" it was intended that the pilot, on the completion of the attack, should abandon the aircraft and land by parachute.

The Me.163 is built around the Walter (T stoff C stoff) rocket unit, Figs. 26 and 27. This unit, which weighs 415 lb., develops a thrust of 3,740 lb. for a fuel consumption of 20.5 lb., the fuel therefore has a specific thrust of about 180 lb./sec. In the particular unit shown in the figure, a small cruising jet has been embodied so as to give the aircraft a longer endurance.

The Me.163, Figs. 28 and 29, is a tailless aircraft, it has a span of 30.5 ft., a length of 18.7 ft. and a gross wing area of 211 sq. ft.

The Me.163C, which is the aircraft fitted with the cruising jet motor, has a take-off weight of 11,300 lb., of which 5,570 lb. is fuel. The aircraft is estimated by the Germans to have a top speed of nearly 600 m.p.h. and it could climb to 40,000 ft. in just over three minutes. In doing this it practically exhausted all its fuel, since at full power the fuel only lasted about 4.5 minutes.

" Natter " the other piloted rocket aircraft was intended to be catapulted at an angle of 75°, by means of two solid fuel rockets, which were to be jettisoned at a height of about 5,000 ft. A T stoff C stoff rocket motor then took over, and the aircraft continued to climb to the desired height; this should have brought the aircraft into the bomber formation and the pilot after completing his attack dived away and abandoned his ship.

The feature of both these rocket-propelled aircraft is their high rate of climb. This was a feature which it was hoped would make it possible to defend a target with little warning of the impending attack, a feature that did not make it necessary for the aircraft to be in the air, except when carrying out an attack; and it was on a consideration of this kind that the very short endurance was justified.

FACTORS AFFECTING ROCKET PERFORMANCE

In discussing rocket performance it is useful to introduce the ratio

$$\frac{\text{Fuel weight}}{\text{Total weight}}$$

which is often referred to as the "α" of the rocket.

The α in the case of the A4 rocket is about 0.69, that is to say, more than two-thirds of the rocket weight is given up to fuel.

EFFECT OF α ON RANGE

Fig. 30 shows the effect of α on range, curves being drawn for a series of rockets of different all-up weights. It will be seen at once that variations of α have much more influence, on range, than variations of size: a 50 per cent. increase in the all-up weight for the same α, only increasing the range by

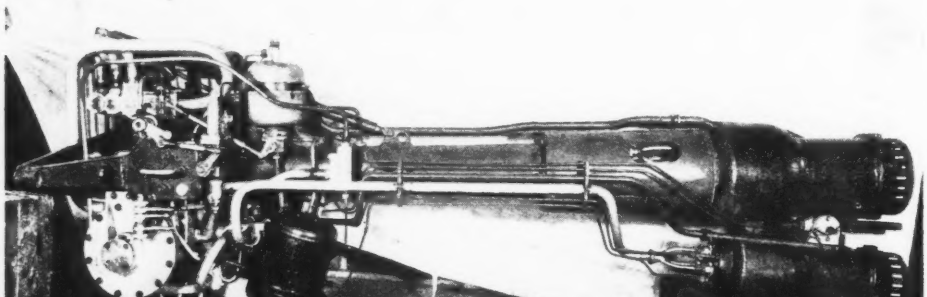


Fig. 26
General View of Walter Rocket Unit.

Fig. 27
Exit Nozzles of Walter Unit.

about 10 per cent., whereas only a 2 per cent. change in $\varkappa$ is required to produce the same effect.

It is interesting to observe that if the 2,150 lb. of warhead, carried by the A4 rocket, were replaced by fuel, $\varkappa$ would have increased to 0.76, and the range would then have been about 350 miles. Since an $\varkappa$ of 0.76 probably approaches the limit of what it is practical to achieve, it is clear that the range of a single stage rocket of the kind we are discussing is fairly limited.

EFFECT OF FUEL ON RANGE

The limitation in range, that we have just considered, at once raises the question of whether some better fuel giving an improvement in specific impulse might not be possible.

Fig. 31 shows the influence of the specific impulse on range; it will be observed that an increase in the S_1 from 215 to 298 would double the range of the rocket. Against this, however, it must be remembered that improved S_1 of a given fuel can only be had at the expense of greatly increased working temperatures and pressures of the gases in the combusion chamber. Practical consideration of the loads coming on the combustion chamber will therefore limit what is possible to achieve in this way. In passing, however, it should be noted that the working S_1 of a rocket fuel does improve with altitude, as a consequence of the increased pressure ratio

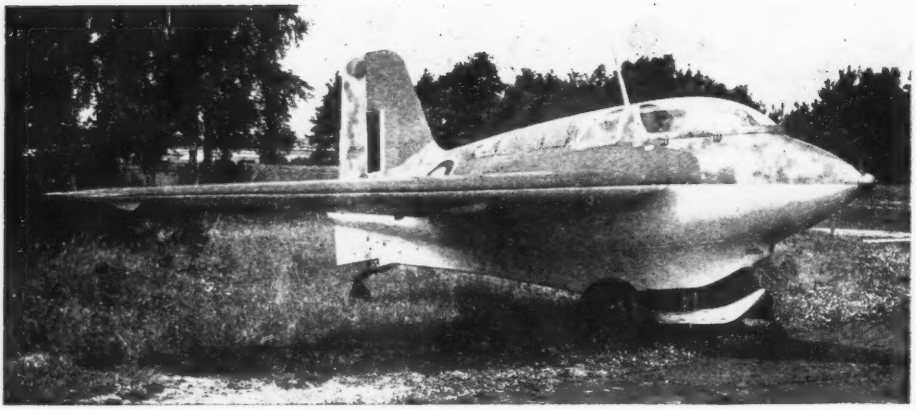


Fig. 28
General View of Me. 163 Rocket Aeroplane.

across the venturi as the atmospheric pressure falls off.

The other course open to the rocket designer to improve range is the choice of different fuels; here the heat of combustion of the fuel and the molecular weight of the resulting products of combustion are the important factors. Table II gives the specific impulse of a number of fuels, calculated on the basis of working conditions corresponding to the venturi characteristics of the A4 rocket.

Most practical value of the S_1 lies between 180-240, but it will be seen from the above table that by using hydrogen and oxygen the value could be pushed up to well over 300 while still retaining a moderate combustion temperature, and some still further improvement in this S_1 would be possible at the expense of the working temperatures and pressures.

VARIATION OF RANGE WITH THRUST AND TIME OF BURNING

The range is also influenced by the rate of burning of the fuel, *i.e.*, on the thrust developed, and therefore on the accelerations experienced during the initial stage of the flight. Fig. 32 shows the effect of variation of burning time on range. It will be observed that the effect is not large, but nevertheless variations of the order of ± 10 per cent. in range are possible, and it is important therefore to take this factor into account when deciding on the venturi and fuel pump design.

TABLE II.

SPECIFIC IMPULSE OF VARIOUS FUELS.

Fuel.	Volume Ratio.	K° Combustion Temperature.	γ	Molecular Weight.	Specific Impulse.
Ethyl Alcohol and Oxygen ..	1:1.15	2,780	1.2	20.0	235
Petrol and Oxygen	1:1.4	2,240	1.3	16.5	224
Hydrogen and Oxygen ..	1:3.2	2,560	1.2	8.4	343
Hydrazine Hydrate and Hydrogen Peroxide	1:2.03	1,750	1.2	24.0	200

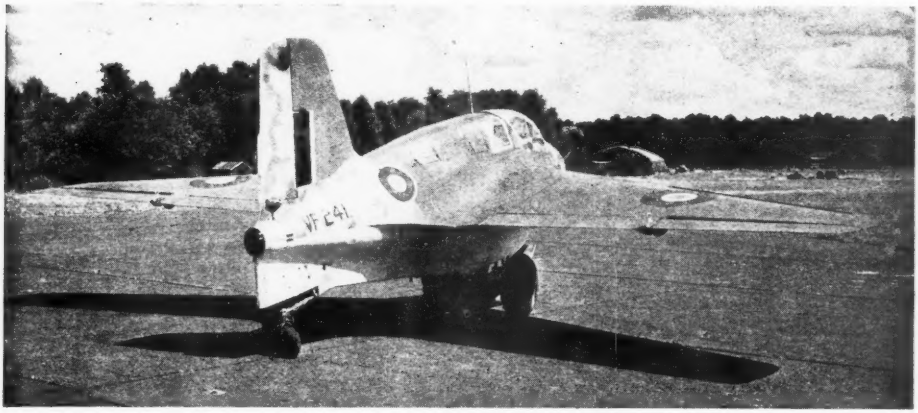


Fig. 29
Me. 163 Rocket Aeroplane.

COMPARISON BETWEEN CONVENTIONAL AIRCRAFT AND ROCKET

It is perhaps as well at this stage to pause and make a brief comparison between a conventional aircraft and the rocket or the rocket propelled aircraft as developed by the Germans. In the following table the main characteristics of a fairly representative high speed bomber are compared with those of the V2, and as a matter of interest the corresponding figures for the V1 are also included.

The item in the weight comparison that attracts attention is the high pay load of the V1, and the low pay load of the V2. As might be expected, the percentage structure weight of the V1 does not differ greatly from that of a conventional aircraft, but the light power plant and the small amount of equipment carried has enabled the pay load to be high. The V2 on the other hand, although it gains on account of the light form of power plant, and a very small structure and equipment weight, is handicapped by the very large amount of fuel that has to be carried.

Another point of particular interest is the fuel consumption figures, expressed in lb. per horse-power hour. The V1 engine is very uneconomical, and the fuel consumption, expressed in this way, is at least five times greater than a more normal petrol engine, but the V2 on the other hand at near all burnt, which is of course the most favourable condition, does achieve a consumption figure that is only twice that of the conventional engine.

TABLE III.

COMPARISON BETWEEN CONVENTIONAL HIGH SPEED BOMBER AND ROCKET.

Item.	Fast Bomber.	V2.	V1.
	Percentage Weights.		
Structure ..	29	13	25
Power Plant ..	24	8	8
Fuel	19	69	19
Equipment ..	13	2	1
Pay Load ..	16	8	46
	Speed in Miles per hour.		
Take-off ..	120	0	200
Maximum ..	400	3,400	350-400
	Other Data.		
Ceiling (ft.) ..	40,000	350,000	9,000
Range (miles)	1,600	220	175
Endurance	6.5 hrs.	5.0 mins.	30 mins.
Fuel Consumption (lb./sec.)	0.57	275	0.66
Fuel Consumption (lb./h.p. hr.)	0.75	1.6 (at near all burnt)	4.3

Table IV makes a further comparison, this time between a typical fighter aircraft and the German Me. 163C.

TABLE IV.

COMPARISON BETWEEN TYPICAL FIGHTER AND ME.163C ROCKET PLANE.

Item.	Typical Fighter.	Me.163C.
	Percentage Weights.	
Structure	30	25
Power Plant	38	9
Fuel	12	49
Equipment	10	4
Pay Load	10	13

Here again the light power plant, and the light equipment weight of the rocket 'plane are offset by the very large amount of fuel that has to be carried, the pay load however has not been sacrificed, and as was pointed out earlier, although the top speed and climb performance of the Me. 163 are high, the endurance is very short.

THE FUTURE

At this point in a lecture of this kind, I think the lecturer is entitled to embark upon a little speculation regarding the future. I have attempted, during the course of the lecture to set up one or two signposts, which give an indication of some of the possibilities and some of the limitations of rocket propulsion.

Ten years ago, when it was only the very fast aeroplane that travelled at 400 miles per hour, we were still speaking of a "sonic barrier," and we were attempting to predict

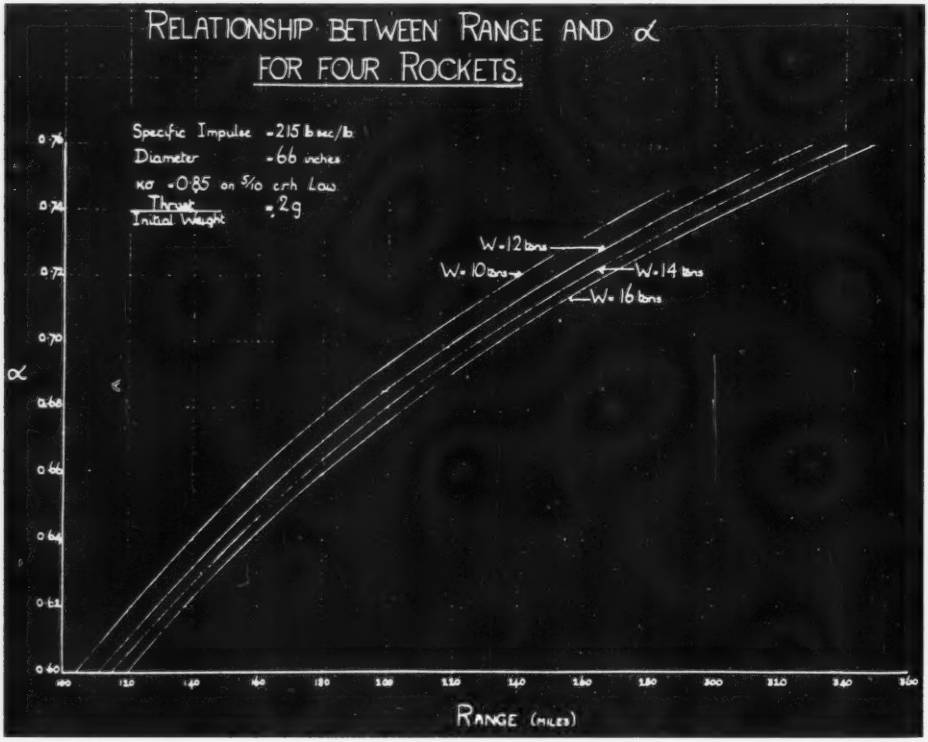


Fig. 30
Effect of X on Range.

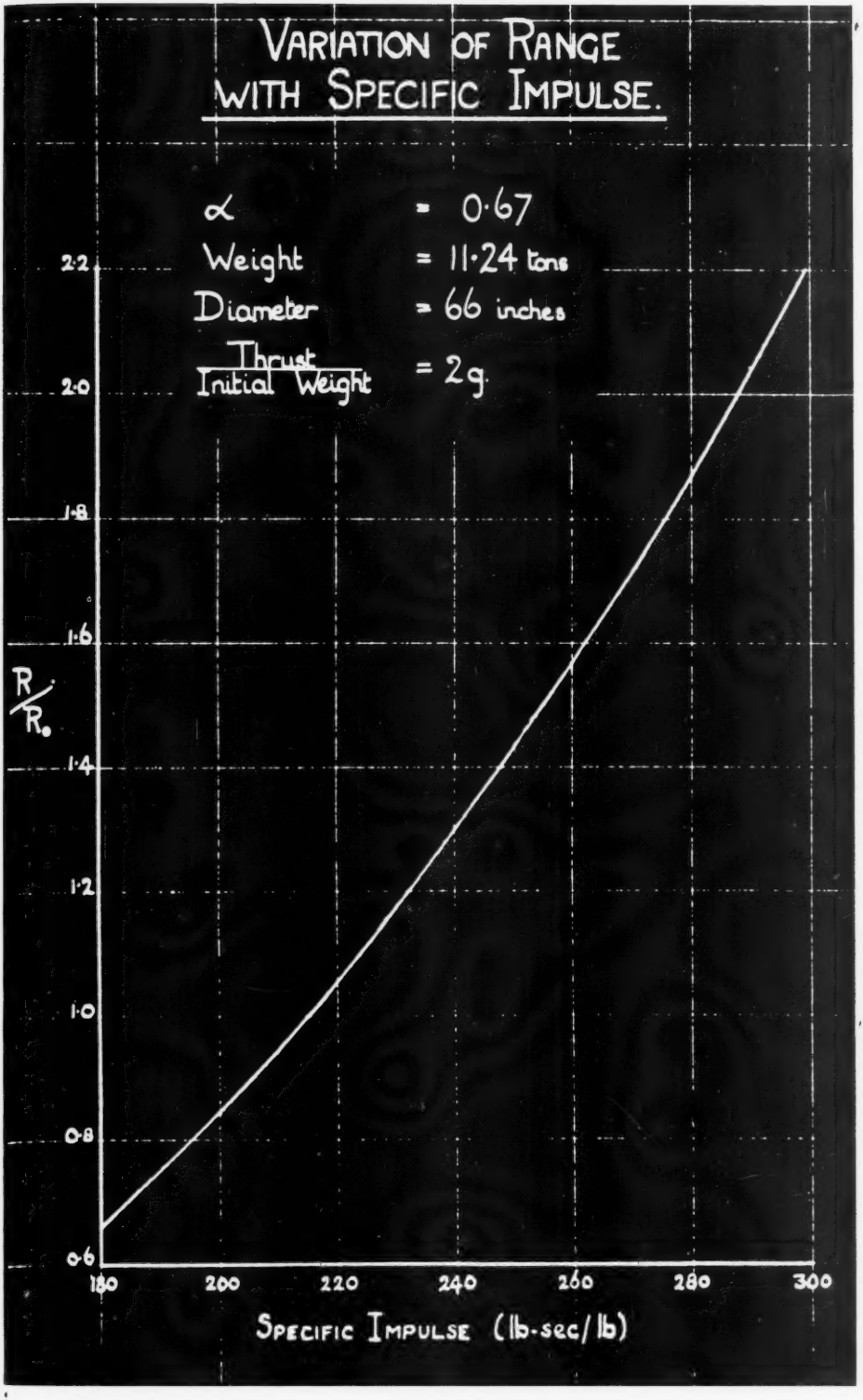

VARIATION OF RANGE WITH SPECIFIC IMPULSE.
α = 0·67
Weight = 11·24 tons
Diameter = 66 inches
Thrust / Initial Weight = 2g
2·2
2·0
1·8
1·6
R/R₀
1·4
1·2
1·0
0·8
0·6
180
200
220
240
260
280
300
SPECIFIC IMPULSE (lb-sec/lb)

Fig. 31

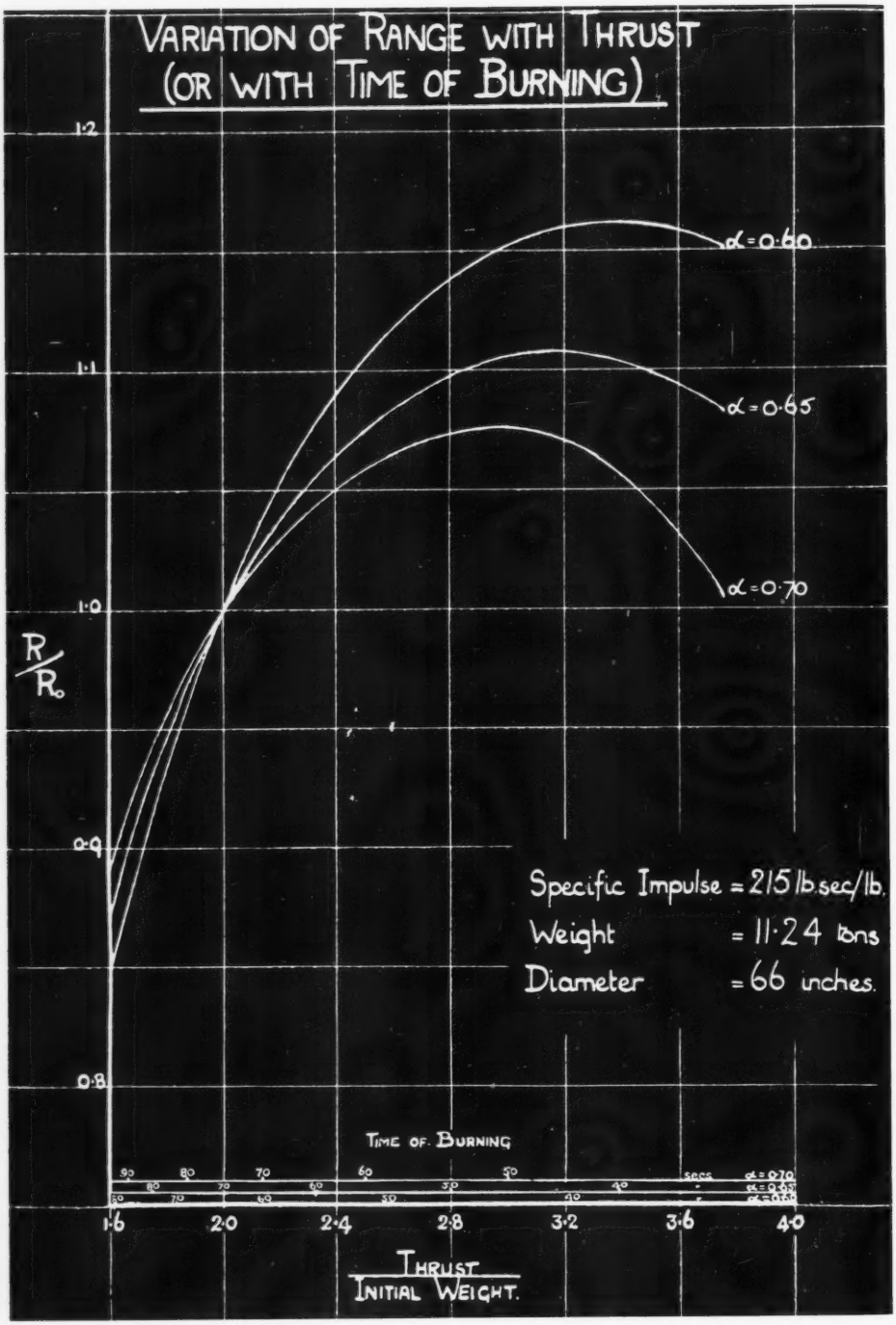

VARIATION OF RANGE WITH THRUST
(OR WITH TIME OF BURNING)
1·2
1·1
1·0
0·9
0·8
R/R₀
α = 0·60
α = 0·65
α = 0·70
Specific Impulse = 215 lb.sec/lb.
Weight = 11·24 tons
Diameter = 66 inches.
TIME OF BURNING
SECS
1·6
2·0
2·4
2·8
3·2
3·6
4·0
THRUST / INITIAL WEIGHT.

Fig. 32

the speed which, so to speak, would mark the limit of man's endeavour.

The rocket motor, with its light weight, and its very large thrust per square foot of frontal area, has now made flight at speeds well into the supersonic range possible and a few calculations have been made, which illustrate some of the possibilities. Starting with the A4 rocket, we have assumed that the Warhead is removed, and replaced by a pressure cabin and pilot, and in addition, the rocket is fitted with wings; the wing area being arranged to provide for a landing wing loading of 35 lb./sq. ft.

Three cases will be touched on briefly. In the first, the A4 rocket with wings is assumed to be fired vertically, and controlled in the same manner as a normal A4 rocket until it reaches the top of its trajectory, when the pilot takes over and glides the rocket to the ground along its optimum glide path.

In the other two cases, the A4 rocket with wings, has been assumed to be taken up by means of a booster rocket rather on the lines of the German A10 project mentioned earlier in the lecture. In this way the rocket reaches a height of 80,000 feet before being released at a speed of 3,000 m.p.h. (these conditions correspondents to the height and speed reached by the A4 rocket with wings at all burnt, when it is fired vertically upwards in the normal way). From this point onwards the rocket with its full fuel continues the flight, and in the first of these boosted cases it has been assumed that the rocket venturi is re-designed, and now provides a thrust corresponding to level flight at 1,640 m.p.h., at 80,000 feet, while in the other the re-designed venturi provides a thrust which enables the rocket to continue the climb. The results of these calculations are set out in Fig. 33, where the trajectories for three cases are compared with the trajectory of the normal A4 rocket without wings.

The first striking thing about the results is the effect that the addition of wings has on range. The range of 180 miles of the normal A4 rocket without wings has now been raised to 350 miles.

Still more striking however is the effect of the booster on the general performance. Ranges of 1,500 to 3,000 miles now appear to be possible, and the advantage of continuing the climb at the end of the booster stage is also very marked. It is interesting to note that with a rocket boosted in this way it would be possible to complete the journey from London to New York in well under the hour.

The first of the two cases of the rocket with wings presents no problem that is outside of the experience already gained by the

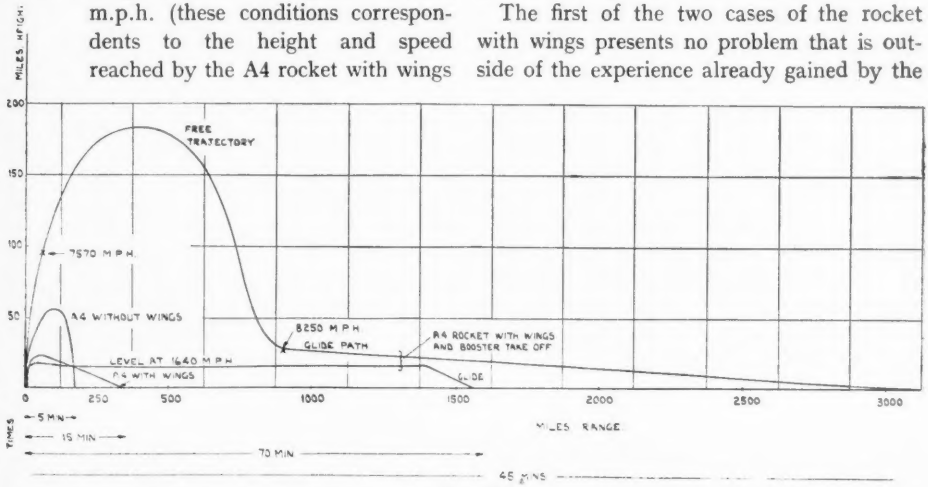


Fig. 33
Trajectories of Rockets with Wings.

Germans during their work on the A4 rocket, the third case however does raise many new problems. It contemplates for example flight at over 8,000 m.p.h.; this means a stagnation temperature of nearly 7,000°K, so that even though radiation may play an important part in keeping the temperature down, the pilot would nevertheless find himself enclosed in a body the skin of which, to say the least of it, was uncomfortably hot. Over a large part of the flight too, the rocket would be moving on a free trajectory, since the wings cannot provide sufficient lifting force to control the motion. Over this part of the trajectory the pilot would be subject to zero g, and it would not be until the rocket returned to a point about 28 miles above the earth's surface that the pilot could begin to assume control of his machine.

The booster contemplated in connection with these schemes would of course be very large; it would certainly weigh about 100 tons, and the Germans in planning their schemes had hoped that the booster after being jettisoned, would be recoverable.

The results of the calculations that have just been outlined are admittedly very sketchy, but they do, I hope, give some indication of the way in which it may be possible to fly at supersonic speed. I do not however want to minimise the difficulties of this problem. As yet nothing, or practically nothing, is known about control at these speeds, nor of the difficulties that will be encountered in passing through the speed of sound. The vertical ascent at the beginning of the trajectory avoids some of these issues, while for control over a greater part of the flight through the upper atmosphere it is quite likely that the controllers will have to be in the form of small rocket jets, since the external forces called into play are far too small to affect the motion. These, however, and the many other problems are matters that will have to be left to the future.

CONCLUDING REMARKS

I should like in conclusion to say how much I am indebted to the many workers in this field who took part in the A4 rocket investigations, on whose work I have drawn quite freely in preparing the present lecture.

I would also like to emphasise that all opinions expressed in this paper are my own and do not necessarily represent the opinions of the R.A.E. or the Ministry of Aircraft Production.

APPENDIX I

Detail Summary of Weights of A4 Rocket Components.

Warhead (with Amatol filling)	2150 lb.
Radio and Instrument Bay.	
Radio compartment including end frame	325 lb.
Radio equipment including mountings	155 lb.
Instruments, electrical equipment and wiring	415 lb.
Compressed nitrogen bottles and fittings	80 lb.
	975 lb.
Tank Bay.	
Shell structure around tanks (including glass wool) ...	1185 lb.
Oxygen tank and fittings ...	375 lb.
Alcohol tank and fittings ...	235 lb.
Liquid Oxygen (tank full) ...	10940 lb.
Alcohol (tank full)	8370 lb.
	21105 lb.
Power Unit Bay.	
Steel structure around power unit	410 lb.
Auxiliary power unit mounting and end frame	260 lb.
Auxiliary power unit	880 lb.
Pipes and valves	70 lb.
Venturi, including burner assembly	1025 lb.
Hydrogen Peroxide (T Stoff)	370 lb.
Sodium permanganate ...	29 lb.
	3044 lb.

Control Surfaces.

Stabilising fins	750 lb.
Internal control vanes, including servo units	470 lb.
External control circuits ...	115 lb.
	1335 lb.

APPENDIX II.

Performance Details of Rocket Propulsion System.

Turbine.

Mean diameter of blades and nozzles	18.5 ins.
Main flow through turbine	3.5 lb./sec.
Working pressure	330 lb./in.^{2}
Turbine back pressure ...	1.6 lb./in.^{2}
Estimated power	680 h.p. at 5,000 r.p.m.
Endurance at 3.5 lb./sec.	113 secs.

Main Fuel Pumps.

Oxygen Pump.

Overall impellor diameter	10.55 ins.
Flow at 5,000 r.p.m. ...	160 lb./sec.
H.P. at 5,000 r.p.m. ...	320
Delivery pressure	350 lb./in.^{2}

Alcohol Pump.

Overall impellor diameter	13.45 ins.
Flow at 5,000 r.p.m. ...	125 lb./sec.
H.P. at 5,000 r.p.m. . .	360
Delivery pressure	370 lb./in.^{2}

Assuming that 5 per cent. of the alcohol is used inefficiently as coolant, and 5 per cent. returned by the spill valve to the lower pressure side of the pump.

Total fuel delivered to combustion chamber	275 lb./sec.
Specific thrust	220 lb./sec.
Total thrust	60,000 lb. (27 tons).
Capacity of main tanks ...	19,300 lb.
Endurance at full flow ...	70 secs. (approx.)

DISCUSSION

THE PRESIDENT: The paper was of great interest and it was fortunate that the war had ended when it did, for the possibilities were rather hazardous to all of them.

SIR ALWYN CROW (Director-General, Projectiles, Ministry of Supply). He complimented Mr. Perring and his staff on their extraordinary job of reconstructing the rocket from a quantity of almost unidentifiable pieces of metal which had resulted from an air burst of a rocket over Sweden. It was an immense technical feat, for they were able, from the fragments, to give the Government practically full details of what was to be expected. That information was needed at the time, for the Government had been concerned with policy, as, for instance, what should be done about evacuating London, and so on. The re-construction was as near right as it could have been.

Since then they had reconstructed, among other things, some of the V2 rockets at Cuxhaven and had fired them. There had been experiments by a combined team of soldiers and British technicians and some of the German technicians had told what they knew.

The work had not been easy for the Germans. The V2 had suffered from the fact that the time interval between the assembly and filling and the launching had had to be cut down in order to ensure a low proportion of failures or, to put it the other way, a high proportion of successes. The Germans had found that, as the time interval was progressively reduced, the number of failures had dropped, ultimately to the order of 1 or 2 per cent. As a result, there was little coming along the production line.

A vast quantity of material, amounting to many train-loads had been collected, but from it they had been able to assemble only a few complete rockets. However, at Cuxhaven they had launched the necessary number successfully, and the information gained had proved very valuable. The trials

were not quite representative, for they had used a control gear of their own instead of the German control gear.

The comments in the paper about the future were extremely interesting. Undoubtedly something of the sort described would have to be expected, and plans were in hand in this country with a view to exploring future long-range projectiles on a fairly large scale. It was a necessary parallel to the atomic bomb development, and it would have to be investigated with the greatest possible care and attention.

PROFESSOR A. A. HALL (Imperial College, Fellow): He emphasised the bravery of the German technicians in tackling the project, and said they should do well to note that the V2 did give evidence of the height and speed achieved by the use of the new fuels and the new motors used.

He reinforced Mr. Perring's reference to control, for although the necessary thrust could be developed, it was not fully efficacious without the proper means of controlling it. There was an immense field for further research on very high speeds.

DR. I. LUBBOCK (Asiatic Petroleum Company): From the remarks of Professor Hall he had gained the impression that a good deal of the development of the V2 by the Germans was novel in fundamental principles. He preferred to say, however, that nobody would deny them the right to credit for a great many ingenious applications of what had been a well-known rocket system. The combustion of liquid oxygen with petrol or alcohol had been done, not only in this country, but also in the United States of America, at about the time the Germans were doing their work on rocket development.

An interesting feature which had enabled the Germans to achieve what appeared to be such advanced results was their military conception of what they had to do to win the war. In other words, the urge was there, and they had had the full financial backing of the military authorities in putting an enormous amount of manpower—scientists and technicians—on to the task of projecting the rocket over the distances involved. In this country they had been well aware that, with $\frac{1}{4}$'s of the magnitude mentioned by Mr. Perring, long ranges could be achieved; but in their case there was not the same urge.

After listening to Mr. Perring some people might perhaps be inclined to assume that a good deal about the design and construction was easy. But those who wished to develop ideas on the subject should realise that the combustion chambers and the process described for forming a flame and obtaining a reaction were not so easy as might appear.

In presenting the paper Mr. Perring had had to condense a great deal of information into a small space; but they should be careful not to jump to the conclusion that they could use other fuels readily. The venturi and the method of cooling had given the Germans a great deal of trouble.

The combustion cups and the venturi and the film cooling represented a system which only just worked. They knew from their experience that the rates of heat transfer in the throat, and the heat in a combustion chamber of the kind used, were fantastic in their magnitude. It was only the film cooling that had enabled the Germans to just get the thing working. The slightest change in the cups or in the mixing of the liquid oxygen and alcohol caused hot spots, and the venturis had frequently burned out.

In addition, the Germans had devoted a great deal of time to the work of overcoming the expansion of the outer jacket of the venturi. It had not occurred to them until late in the day to use the loops to overcome expansion. Before that they had experienced the tearing of the casing at the weld, as the result of which alcohol had leaked by and there were explosions as the result of that leakage into the outer casing. The reason

for the explosion was not quite clear; the Germans themselves were not clear about it. But there were gland leaks from the pumps, and the Germans were fairly well convinced throughout that any mixture of alcohol vapour and oxygen in that state was liable to be fired fairly readily.

In this country they had been rather puzzled as to the reason for lagging the steel pipe carrying the hot gases to the turbine nozzles. One reason was to conserve the heat; another was to guard against the hot zones being formed in the space above the combustion chamber, which might lead to explosions.

As Mr. Perring had pointed out, the problem of skin temperature was rendered difficult due to stagnation temperature. A great deal of heated argument had raged around the stagnation temperature, and as to why, with the calculated 1100°C., there was only in effect about 600°C. It had been suggested that one method of calculation would be to take the work done by the projectile against the head resistance in the descent through the dense atmosphere, and to try apportionment as between the projectile and the heat given to the air, taking into account also the re-radiation.

He had made a few tentative calculations, which had led him to the rather curious result that the radiation outwards from the temperature built up on the skin was quite small. It would not account for the heat loss, unless higher temperatures were assumed than were actually reached. The reasons for the temperature stabilising at 600°C. instead of 1100°C. were to be looked for in conduction into the shell, radiation across the venturi to the tank, and chiefly the frictional drag.

Considering the air bursts that had occurred, they might assume that the skin had reached the temperature of 600°C.; as the heat transfer would be of the order of 250 B.Th.U. per second, they could appreciate that the radiation in the interspace between the outer skin and, say, the alcohol tank was quite high, and that in the period during which the rocket was passing down into the dense air the alcohol tank could readily be heated to temperatures very nearly approaching that. If what the Germans had done with the alcohol tank was examined, they realised that a condition was created which was liable to lead to explosion. In other words, the alcohol tank was pressurised with nitrogen and was then subjected to ram compression; the ram compression was shut off after about 40 seconds, and the vessel was then pressurised with nitrogen. So that, when the rocket was in the stratosphere and above it, the alcohol tank contained air and nitrogen and alcohol vapour. Then the rocket careered down through the dense air and became very nicely heated. In those conditions it was not surprising that there were bursts in the air, which did not cause so much damage as the bursts which had occurred on the ground.

Those facts indicated what happened when projectiles were brought down through the dense air at nearly 5,000 ft. per second. If rockets descending at a speed of 6,000 or 7,000 ft. per second were visualised, it could be appreciated that the stagnation temperatures would render the position hopeless from the point of view of the survival of the rockets so that they would do damage.

Thus, as Mr. Perring had said, they could not continue to increase the range of a free rocket. Everything that was done to increase impulse and α merely gave higher velocity and the rocket would enter the dense air at equivalent high velocity and the stagnation temperature would cause it to explode. Therefore, Mr. Perring had concluded—and the Germans had thought along the same lines—that if they wanted fantastic ranges they must do something else.

Mr. Perring's proposals were intensely interesting. He would not care to comment on them unless he had been able to sit down, with six or eight mathematicians, and work

on each scheme for several months, making exact calculations.

He had been deeply impressed by the work which Mr. Perring had done during the period when they in this country were trying to assess what the rockets were likely to do. It was a job of which the R.A.E. could justly be proud.

MR. D. R. NEWMAN (de Havilland, Ltd., Associate Fellow): He was particularly interested in Mr. Perring's diagram of the possibilities of the rocket when fitted with wings. People spoke from time to time of the possibilities of cruising in sustained flight at heights of 50 or 100 miles and at speeds of 5,000—10,000 m.p.h., at very little cost. But it was not often remembered that, in order to achieve sustained flight in those conditions, it would be necessary to have an impractically low wing loading, the point being that there were certain maximum values which could not be cheated by flying in the less dense air.

On Fig. 33, which showed the trajectories of rockets with wings, he had been measuring the gliding angles for different ranges. For ranges between 350 and 1,500 miles the gliding angle would be probably 1 in 10; for a range of 3,000 miles it would be 1 in 75, or thereabouts. The rocket would be slowing up throughout the glide, and the speed would be down to about 250 m.p.h. at sea level. Could Mr. Perring indicate what would be the average L/D ratio of the rocket during its flight?

MR. PRIBRAM (Royal Aeronautical Society, Associate Fellow): He was interested in the engineering aspect of the rocket as a simple heat engine and would like to have had a little more information on that. For example, how were the liquids making up the fuel mixed, so that there should be no free alcohol or oxygen left? From internal combustion engine design, they knew that a high degree of turbulence was required to obtain a uniform mixture, and he supposed that the brass roses were designed for that job, although that was not stated in the paper. With a flow of 275 lb. of fuel per second that must have been a formidable problem, and he had been told that at some stage buffer plates had been fitted to the cups, presumably to prevent solid sprays of fuel from reaching the combustion chamber. He also thought that the injection of alcohol directly into the venturi was at least partly intended to insure that no free oxygen reached the internal controllers, which were of hard carbon and would otherwise be quickly burned.

He did not understand the remark about the fuel being burned initially "without shock." If this referred to conditions in the combustion chamber, there was no shock wave to be expected in it; possibly what was meant was that there was no thrust?

He questioned the accuracy of the diagrams on Fig. 33; especially the straight glide path of the A4 rocket. It seemed incredible that the conditions for optimum gliding angle should remain unchanged from a speed of 8,250 m.p.h. down to well below the speed of sound.

It seemed to him that the right way to achieve a greater range lay, not necessarily in building bigger and better rockets. The efficiency, in terms of useful load, of the A4 was already very low and when the A10 with booster was considered it became fantastically low. Surely a better way would be to utilise the rocket drive where it was really advantageous, *i.e.*, at extra-atmospheric heights, but take the rocket up to that stage and start it by some other method less wasteful of fuel. Various possible combinations presented themselves, as, *e.g.*, an athodyn which could be used as a rocket chamber when the atmospheric oxygen gave out.

He would refer to a suggestion, made, he thought, by the German technicians, to launch the rocket from a jet-propelled parent plane at a height of, say, 70,000 ft. and a

speed of 600 m.p.h. This was, in principle, not different from the launching of rocket projectiles now in use; surely that was a more economical process than to launch a 100-ton rocket with a useful load of just over 1 per cent. and a recoverable first stage, which would presumably fall back on to the population of the country from which it was launched ?

DR. W. F. HILTON (National Physical Laboratory, Aero Division, Associate Fellow): He urged that "stagnation point" should be referred to rather than "stagnation temperature." Usually the recorded value was only about 90 per cent. of the calculated value.

Had Mr. Perring endeavoured to obtain information on the value of the stagnation point of the skin for temperatures of about 170° ? He suggested that the theoretical temperatures might not represent anything like stagnation.

He objected to the use of the word " venturi " to describe the nozzle used in the V2, as in a venturi the maximum speed was attained in the throat, whereas the nozzle of the V2 and similar rockets showed constant acceleration throughout the axis.

MR. S. SCOTT-HALL (Ministry of Supply, Fellow): How far were they at the present stage from being able to fly outside the earth's gravitational field?

MR. J. BLACK (de Havilland Aircraft Co., Ltd., Student): Were the external control surfaces meant to be of value only during part of the flight of the projectile ? It seemed that after the rocket became supersonic they would be comparatively ineffective.

WING-COMMANDER H. M. YEATMAN (Associate Fellow): He referred to Fig. 33, dealing with the trajectories of rockets with wings, and asked what was the amount of g to which the pilot would be subjected when flattening out on the glide. Unless his radius of curvature were more than 100 miles, apparently he would be subjected to about 9 or 10 g.

MR. R. C. G. SLAZENGER: Were the L/D ratios obtainable at supersonic speed comparable to those at subsonic speed, and what actually happened at supersonic speed ?

MR. C. F. HODSON (Fairey Aviation Co., Ltd., Associate Fellow): He believed the speed over the main part of the run was said to be something like 3,000 m.p.h., and that reasonable landing speeds were proposed. It was also suggested in the paper that the rockets operated through an angle of about 2°. Assuming an angle of 2° and a speed of 3,000 m.p.h., what was the load on the wings at a reasonable landing speed ?

MR. A. V. CLEAVER (de Havilland Aircraft Co., Ltd., Associate) (*contributed*): One of the most interesting engineering features of the German rocket motors, both of V2 and the Me.163, was the designers' clever solution of the fuel feed problem by the use of turbine driven centrifugal pumps. Was this entirely satisfactory in the view of the German experts, who should be best qualified to pass judgment ? If not, had they any alternative ideas, since it would seem that the fuel feed problem might be one of increasing difficulty on larger sizes of rocket motor ?

Mr. Perring's informative paper broke such new ground that any aeronautical engineer might be forgiven for continuing to ask questions almost indefinitely, even if some of them were of a rather elementary nature. Reliable published data on rocket propulsion had been so scarce that some uncertainty regarding even fundamental points must exist in the minds of many people in the Industry, and Mr. Perring was to be congratulated on making a beginning on the task of their enlightenment. It was to be hoped that further official releases of information would follow.

A number of such points arose in connection with Table II of the paper, and the Author's discussion of it, with reference to future possible developments. He stated there that the specific impulse of an ethyl alcohol/oxygen rocket fuel was 235 lb./lb./sec., but elsewhere in the written paper quoted it as 220, without explaining the discrepancy. While delivering the lecture, however, he did refer to 235 as the theoretical and 220 as the actually obtained values.

Was this a slip, since Mr. Perring's expression: $S_1 = \frac{1}{g}\sqrt{\frac{2\gamma}{\gamma-1}}\frac{RT_o}{M}$ calculated (using the values of combustion temp.. γ, and molecular wt. quoted in Table II) a theoretical value of approximately 380, not 235, unless his arithmetic was at fault ?

Regarding the problem from another angle, one could write $S_1 = \sqrt{\frac{2L}{g}}$, where L was the calorific value of the alcohol/oxygen mixture in ft.-lb./lb. From text book values of L, one would thus estimate an S_1 of about 430 for pure ethyl alcohol and oxygen mixed in the theoretically correct ratio for complete combustion. However, V2 operated on a 75 per cent. water-alcohol mixture, and even then used, for cooling purposes, an excess of alcohol over the weight which could be burned completely in the oxygen available. These factors would result in a reduction of S_1 from about 430 to 380, agreeing with the figure calculated above from Mr. Perring's equation, but not with his quoted 235 in Table II. Similar comments applied to the data given in that table for other fuels.

Assuming that the specific impulse of 235 was the value actually obtained for some condition (perhaps at high altitude), there would then theoretically appear to be room for improvement up to a value of between 380 and 430, still using the same fuel. The 235 value corresponded to a thermal efficiency, $= \left(\frac{S_1 \text{ actual}}{S_1 \text{ theoretical}}\right)^2$, of only some 30 per cent. In a very simple heat engine such as a rocket motor, working between such wide temperature limits, one would have thought this might be considerably bettered. What were Mr. Perring's views on the possibility of improving the 30 per cent. efficiency by improving the component efficiencies of the motor (combustion η, nozzle η, etc.) ? Was there any complex chemical effect of dissociation occurring at the high combustion temperatures which prevented the text book calorific value being realised ?

If the obstacle to using higher specific impulses from a given fuel was the stresses arising from higher combustion temperatures and pressures, then surely some improvement on the welded mild steel V2 chamber should be fairly easy. As regards other fuels such as a hydrogen/oxygen mixture, presumably the consideration of bulk (greater size of rocket and tankage wt.) had to be weighed carefully against possible increases in specific impulse. This point was partly bound up with the question of how important aerodynamic drag was in determining rocket performance; incidentally, what was the difference in form drag coefficient with the propulsive jet on and off, at a given Mach No. ?

Mr. Perring gave them " one or two signposts " which would have seemed startling enough indications to most people before the war. He omitted, however, to direct them right over the cliff, and point out the most fascinating possibility of all those raised by rocket flight, namely that of interplanetary travel. He believed von Braun himself had been less cautious, and had made his own interest and belief in this quite clear.

If Mr. Perring's caution arose from deference to the now eminently respectable R.Ae.S., it was a pity, because the R.Ae.S. should not be allowed to forget that half a century ago its pioneering members were generally regarded as pursuing an end no less revolutionary than astronautics might seem

to-day. Aeronautics also was then considered to be either (a) impossible, or (b) not worth achieving, but time tended to dispose of both such objections.

THE PRESIDENT: Within whose province lay the development of rocket projectiles? That matter raised an interesting point. From the point of view of one who was interested in aircraft, it seemed a proper sphere for the aircraft engineer and aircraft designer, that the knowledge in regard to the best forms of fins, and wings if they were to be fitted, must lie mainly within the range of those connected with the Aircraft Industry. On the other hand, he could well imagine that the people who were interested in projectiles might claim that the effects of wings and so forth on the flight path and on the general problem of the ballistics of the projectiles were matters on which they had more knowledge than had the aircraft designer or engineer. He wondered what Mr. Perring had to say on the subject; probably his solution would be that they should be developed at the R.A.E.!

He would like to know how the Germans had proposed to launch their rocket after it had been taken up to a height by a piloted aircraft. It would not be a pleasant occupation to fly the aeroplane which was to take it up, if the ratio of failures to successes were so high as had been indicated.

MR. PERRING'S REPLY TO THE DISCUSSION

He thanked Sir Alwyn Crow and Mr. Lubbock for their kind remarks about the work at the R.A.E. The reconstruction of the exploded rocket was a team job which the R.A.E. had shared with Sir Alwyn Crow and Mr. Lubbock.

Professor Hall. He agreed with his remarks concerning the bravery of the German technicians in tackling a job of the nature and magnitude entailed in the development of the rocket.

Mr. Lubbock. He was sorry to have given him the impression that the job was easy; he agreed entirely with him about the difficulties that had had to be faced and overcome. No one who had worked in the experimental field for so long as he had done would minimise those difficulties. The German conception was a bold one; in a period of two years they had done remarkable work in overcoming most of the difficulties and developing the rocket to the stage it had reached. The rocket was produced as the result of tremendous effort, fully backed by the German military machine; nothing was spared in the way of experimental equipment, men and resources in order that the job might be done. A detailed lecture on the development of the V2 would make a very interesting story; it would be full of instances showing how, as the various problems arose a solution was sought and a remedy found.

As to the risks of explosion, he had felt all along that the use of the ram pressure as a means of pressurising the alcohol tank was one of the weaknesses of the rocket. He had never been able to appreciate why the Germans had used that method, especially when it was remembered that the explosion temperature of air/alcohol mixtures was about 450°C.; the introduction of such a mixture in conditions where temperatures up to 600° were reached was inviting trouble. It would have been far simpler to carry a few more bottles of nitrogen for pressurising purposes. He believed that it was their experience that most of the air bursts that did occur were due to the bursting of the alcohol tank.

Dr. Hilton. The skin would not be expected to reach the full stagnation temperature but only about 85 per cent. of this value. Conduction and the heat capacity of the rocket, radiation and the change in conditions in the upper atmosphere were all likely to play some part in reducing the skin temperature. Mr. Nonweiller, a member of

the staff at Farnborough, had made a very full analysis of the temperature conditions on the A2 rocket and had concluded that the temperature of 900° absolute which was observed was reasonable. A big factor in reducing the temperature below the stagnation value was the time lag due to the heat capacity of the rocket.

The fuel system had been fairly fully described in the body of the paper, but, briefly, since the pressure in the combustion chamber was roughly twenty times that of the outside atmosphere, it must at ground level be about 300 lb. per sq. in., and in order to maintain this pressure 270 lb. of fuel per second had to be fed into the combustion chamber. The pressure losses in the pipes and pump system, and in the nozzles and jackets, accounted for a pressure drop of about 50 lb. per sq. in. The pressure on the delivery side of the pumps, therefore, had to be approximately 350 lb. per sq. in.

The fuel was delivered to the combustion chamber as a liquid and was burnt after mixing; due to the very high rate at which the alcohol was fed through the cooling jackets, the heat transfer to the alcohol was not sufficient to evaporate the alcohol under the pressure conditions existing in the jackets.

In view of the low rate of fuel flow to the combustion chamber there was no reason to expect a violent explosion.

Mr. Pribram. The fuel mixing was accomplished in the 18 cups formed on the end plate of the combustion chamber. The oxygen was sprayed into the cup from a centrally-placed brass rose fitted into the base of the cup, while the alcohol was fed in radially from a number of small nozzles set out around the side. Since the fuel mixture provided for a very considerable amount of free oxygen after all the alcohol was burned, the mixing was less critical than might otherwise have been the case. The excess of oxygen was provided to maintain reasonable working temperatures in the combustion space and the diffuser.

It was agreed that the problem of handling 250 lb. of fuel per second was not an easy one, but the simple mixing cup arrangements provided by the Germans appeared to be adequate, and no steps were taken to introduce extra turbulence by buffer plates or similar devices. The alcohol sprayed in around the sides of the combustion chamber and the diffuser was, as explained in the paper, primarily for cooling, although it might have played a part in stabilising the flow through the nozzle over the wide range of working conditions it had to operate. Since the gases issuing from the jet were still very rich in oxygen it was very doubtful whether the alcohol could have any protective influence on the carbon controllers.

The reference on page 497 to the fuel being burned initially " without shock " was meant to imply that the combustion was steady; little or no thrust, of course, was built up during this initial stage.

The glide path shown in Fig. 33 would be practically straight, since there would be very little change of incidence in reducing the speed from 8,000 miles an hour down to just below the speed of sound.

He agreed entirely regarding the inefficiency of rocket systems and the real advantage of the rocket was in operations at extra atmospheric heights. For conditions nearer the ground, the athodyn would clearly have advantages.

Mr. Scott-Hall. In order to travel outside the earth's gravitational field it would be necessary to achieve speeds of the order of 24,000 m.p.h. The speed attained by the two-stage rocket was only of the order of 8,000 m.p.h.; so that one had to move a good deal faster than that to get outside the earth's field. To travel at 24,000 m.p.h. would require a three-stage rocket and then by dropping off first one stage and then the other, and of course assuming that materials could withstand the conditions, it should be

possible to obtain the speed to get away from the earth's field and travel as a heavenly body.

Mr. Black. In the body of the paper he had given in some detail his views regarding the external controllers on the rocket. They had always been to some extent a puzzle to himself and his colleagues, because at the take-off they provided no control at all, and again, near the all-burnt condition the control provided by the external controllers was negligible. It was only at an intermediate height that their control value approached that of the internal controllers, but at no stage during the initial part of the trajectory were the internal controllers inadequate; they were quite large enough to provide all the control necessary. Therefore, it was difficult to understand why the Germans had gone to the complication of fitting the external controllers at all.

Wing-Commander Yeatman. The trajectories shown in Fig. 33 were only intended to give some idea of the ranges possible by various methods, *i.e.*, by fitting wings or by two-stage arrangements. They were also not necessarily the optimum ranges for the conditions assumed and it was recognised that there were several issues that might prevent such ranges being practicable. The skin temperature, for example, or the acceleration experienced might rule out some of the arrangements discussed.

In the case considered in the paper the g during the pull-out of the boosted scheme was of the order of 25, an acceptance of a lower g would have shortened the range, though the acceptance of a higher skin temperature and a more nominal value of g for the pull-out would still have made it possible to have achieved a 3,000 miles range.

Mr. Slazenger. The $\frac{L}{D}$ was dependent on the body wing interference, but with good design, values of 5 or 6 should be obtainable.

Mr. Hodson. The angular change of 2° referred to in the paper related to the A4 rocket without wings; and it was unlikely that the rocket with wings would experience such large angular changes.

Mr. Cleaver. The specified impulse values in Table II were correct and he was afraid it was the neglect of the term $(1-\frac{p_x}{\rho_o})^{\frac{\gamma}{\gamma-1}}$ from the velocity relationship that had brought about the difference in the values he had calculated and those given in Table II. Because of the very low values of γ this term became important and could not be neglected in any calculation of specific impulse.

The President. He preferred not to rise to his bait on the question as to who should make rocket aeroplanes. If such development were to be a success, there was room for everyone in the field, for there were so many problems ahead.

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THE DESIGN AND PRODUCTION OF JAPANESE MILITARY AIRCRAFT

by

ALEX P. WEST, A.F.R.Ae.S.

Mr. West joined the de Havilland Aircraft Company, Ltd., as a student apprentice in 1930. *In* 1933 *he transferred to the Bristol Aeroplane Company and remained with that Company until* 1940, *obtaining experience in the Wind Tunnel, Drawing Office and Stress Departments. He obtained his civilian pilot's "A" licence in* 1938 *and was elected an Associate Fellow of the Society in the same year. From* 1940 *to the present time, except for a period of seven months when he was on loan to the British Purchasing Commission in New York, Mr. West has been in Australia as a senior aeronautical engineer with the Commonwealth of Australia, Department of Aircraft Production.*

1. INTRODUCTION

IN JANUARY, 1946, the writer was fortunate in being chosen as a member of a Scientific Mission sent to Japan by the Australian Commonwealth Government. As a member of this Mission he was required to investigate and report on the standard of design of military aircraft being produced by the Japanese at the end of the war.

The time spent in Japan was limited to three weeks, and hence no attempt was made to deal with specific types of aircraft, attention being concentrated on obtaining as much general information as possible on design trends.

These notes deal primarily with airframe design, but some information is given on various allied subjects, such as the value of Japanese research establishments to aircraft designers, and it is hoped that they cover a sufficiently wide field to give a general indication of the efficiency of the Japanese Aircraft Industry.

Appendix I contains detailed information on metallic materials used in the design of Japanese aircraft, together with their chemical composition, physical properties and recommended heat treatment; also, organisation charts of a typical Japanese aircraft establishment.

The writer wishes to thank the Commonwealth Government, Department of Aircraft Production, for permission to publish these notes, and accepts full responsibility for the opinions expressed therein; they do not necessarily represent the official opinion of the Department of Aircraft Production.

2. CONDITIONS IN JAPAN

Upon arriving in Tokyo the Australian Scientific Mission were given access to two valuable sources of information compiled by

the American Forces, and the writer gratefully acknowledges this courtesy; the one source of information was a set of records prepared by the American Air Target Intelligence Group, and the other a series of reports by the American Air Technical Intelligence Group.

When assessing the value of information obtained from interrogation of Japanese technicians and inspection of aircraft factories, due consideration has to be given to the following factors:—

(a) It is extremely difficult to assess whether a Japanese technician is giving complete information or not, particularly when interrogation has to be done through an interpreter; however, allied technicians in Japan were in general agreement that the majority of Japanese interrogated answered questions to the best of their ability.

(b) It is not easy to get a clear picture of the operating conditions which prevailed in Japanese aircraft factories during the war, due to bomb damage, and the frantic last-minute attempts which were made to remove valuable equipment to underground dispersal areas. Vandalism, and the enthusiastic carrying out of an Allied instruction calling for the destruction of military equipment, further added to the difficulties of accurately assessing Japanese production capabilities.

3. OPERATIONAL REQUIREMENTS FOR JAPANESE MILITARY AIRCRAFT

3.1. *Types*

The Japanese considered it necessary to have six main types of aircraft for combat purposes. These types are listed below, and the most recent aircraft for which performance data is available are given as examples of the standard which the Japanese achieved in the respective categories.

(a) DAY FIGHTERS

Example ... ... SHIDEN. Model Type No. ... . NIK2J.
Manufacturer... Kawanishi Aircraft Co.
Engine... ... ... Nakajima H.A.45 M0021.
H.P. at S.L. ... ... 1,990 h.p.
Rated Power ... ... 1,625 h.p.
Rated Altitude ... 20,013 ft.
Top Speed ... 370 m.p.h. at 18,372 ft.
Armament... ... 4 x 20 m/m. wing guns.

(b) NIGHT FIGHTERS

Example ... ... DENKO. Model Type No. ... ... S.1.A.1.
Manufacturer... Aichi Aircraft Co.
Engines ... ... Two Nakajima H.A. 45, Mk. 22.
H.P. at S.L. ... ... 2,000 h.p. each
Rated Power ... ... 1,700 h.p.
Rated Altitude ... 19,685 ft.
Top Speed ... 366 m.p.h. at 22,966 ft.
Armament... ... 2 x 30 m/m. wing guns.
2 x 20 m/m. fuselage guns.

(NOTE.—These figures are calculated as this aircraft was not completed.)

(c) CARRIER AIRCRAFT

Example ... ... JUDY.
Manufacturer... Yokosuka Naval Arsenal.
Engine... ... ... Aichi Atsuta Mk. 21.
H.P. ... ... 1,160 h.p.
Top Speed ... 325 m.p.h.
Armament... ... Two or three 7.7 m/m. guns.

(d) DAY BOMBER

Example ... ... FRANCES.
Manufacturer... Yokosuka Naval Arsenal.
Engines ... ... Two Nakajima Homare, Mk. II.
H.P. ... ... 1,800 each.
Top Speed ... 355 m.p.h.

(e) LONG RANGE BOMBERS

Example ... ... FUGAKU.

This aircraft was only on the drawing boards and suitable engines had not been fully developed; it was, however, the type which the Japanese considered necessary to enable them to bomb America.

Weight ... ... 220,000 lb.
Range ... ... 10,000 miles at 42,600 ft.
No. of Engines 6.
Crew ... ... 7.

(f) KAMIKAZE

Example	OKA 22.
Manufacturer...	Yokosuka Naval Arsenal.
Engines	Jet.
Range	81 miles.
Top Speed ...	298 m.p.h.

3.2. *Range*

The Japanese aircraft designers were required to meet the following range requirements in the various aircraft categories listed.

AIRCRAFT CATEGORY	RANGE REQUIREMENTS
(a) Offensive	30 mins. full power. $2\frac{1}{2}$ hours cruising (normal tankage). 5 hours cruising (with belly tank).
(b) Interceptor	30 mins. full power. 2 hours cruising (normal tankage). 4 hours cruising (with belly tank).
(c) Light Bomber	2,500 nautical miles (normal tankage). 3,000 nautical miles (overload).
(d) Heavy Bomber	2,500 nautical miles (normal tankage). 4,000 nautical miles (overload).

3.3. *Armament*

From experience gained during the war the Japanese came to the conclusion that the following armament was desirable.

AIRCRAFT CATEGORY	ARMAMENT REQUIRED
(a) Offensive	4—20 m/m. 200 rounds per gun.
(b) Interceptor	4—30 m/m. 70 — 90 rounds per gun.
(c) Night Fighter	2—20 m/m. 100 rounds per gun. 2—20 m/m. inclined guns 200 rounds per gun. OR 2—30 m/m. inclined guns 200 rounds per gun.
(d) Light Bomber	1—13 m/m. manual control forward turret 300 rounds. 2—13 m/m. top turrets 400 rounds. OR 1—20 m/m. top turret 250 rounds per gun.
(e) Heavy Bomber	2—13 m/m. manual control forward turret 400 rounds per gun. 2—13 m/m. side guns 300 rounds per gun. 2—20 m/m. top turret 600 rounds per gun. 2—20 m/m. bottom turret 400 rounds per gun. 2—20 m/m. tail turret 600 rounds per gun.

4. DESIGN REQUIREMENTS FOR JAPANESE MILITARY AIRCRAFT

4.1. *Basis of Stress Analysis*

A simplified form of V-n diagram is in general use; in some cases it is based on flight test results and in others on purely theoretical data. In the latter case acceleration is plotted against dynamic pressure (*i.e.*, $\frac{1}{2}\rho v^2$), and a straight line relationship is assumed to denote changes in " n " with change in dynamic pressure.

4.2. *Limit Load Factors*

The following load factor ranges are laid down and the actual value of the load factor to be used for any specific aircraft is fixed after discussions between the designer of the aircraft and the military research establishment.

AIRCRAFT TYPE	LOAD FACTOR RANGE
Large	Between 2.0 and 3.0
Medium	Between 2.5 and 3.5
Carrier	Between 3.0 and 5.0
Dive Bomber ...	Between 5.0 and 6.0
Kamikaze ...	5.0
Fighters	Between 6.0 and 7.0

4.3. *Factors of Safety*

The appropriate limit load factor is multiplied by 1.35 to obtain the design yield load and by 1.80 to obtain the design ultimate load for stress analysis.

4.4. *Materials Used and Permissible Stresses*

In general the materials used in the airframes of Japanese aircraft are similar to those in common use in England and America. In Appendix I will be found a detailed list of materials used, design stresses, chemical composition and recommended heat treatments. This information was obtained from Commander Otsuki, formerly of the Japanese Imperial Navy and Chief Technical Officer at the Yokosuka Naval Arsenal. It will be noted that conforming to American practice the 0.2 per cent proof stress is used in calculations relating to design yield strength.

Magnesium alloys were not used by the Japanese to any great extent due primarily to limited availability of these alloys; in fact, at one stage of the war supplies were so limited that castings designed in magnesium alloy had to be produced in aluminium alloy.

4.5. *Stressing Methods*

It will be appreciated that to cover this subject with any degree of detail in the time available was impracticable, particularly when it is remembered that all calculations are in the Japanese language.

However, the Japanese designers are familiar with such methods as Wagner's analysis of tension field spars and with strain energy methods of dealing with redundant structures.

An examination was made of the Japanese equivalent to English standard design data sheets, and it was concluded that the data sheets issued in Japanese design offices were very similar, and in several cases identical, to those used by the Allies.

4.6. *Special Design Factors*

The desirability of taking special precautions when stressing particularly important parts of the airframe is appreciated by Japanese designers. For example, when stressing control hinges an additional factor of 1.20 is used. For members subject to alternating stress or fatigue an additional factor of 1.44 is used, and castings have a factor of 2.0.

No definite factors are laid down to safeguard against stress concentrations resulting from rapid change of cross section or eccentrically applied load, but at the discretion of the designer responsible, permissible stresses are reduced by an amount ranging from 15 per cent to 30 per cent according to the importance of the part being stressed and the severity of the stress concentration.

4.7. *Mechanical Testing*

It is normal practice in Japan to test complete wings to destruction in order to check the accuracy of design calculations, but the writer was surprised to note that in some laboratories shot bags were still being used for this purpose, as for example, at the Aeronautical Research Institute of the Imperial University of Tokyo. At the first naval technical arsenal, Yokosuka, a more modern test rig using hydraulic jacks had been installed. The latter test rig was capable of being used for fatigue testing and had a loading time cycle of approximately one minute.

A novel type of undercarriage test rig was being installed at Yokosuka. This rig launched the undercarriage down an inclined ramp, thus more accurately representing landing conditions than the more usual type of test rig which drops the undercarriage in a vertical plane (see Section 6).

5. DEVELOPMENT TRENDS IN JAPANESE AIRCRAFT DESIGN

5.1. *Armament*

In Section 3.3, particulars were given regarding the armament which the Japanese military authorities considered necessary. At

the end of the war a 60 m m. gun and a 75 m m. gun were being developed, but even if the war had continued it is doubtful if these guns would have been produced, as the success which the Allies were having with rockets had decided the Japanese to develop the latter weapon.

Up to as recently as 1944 the Japanese still had a considerable number of manually-operated gun turrets, and some were still in use at the end of the war. Both electrically and hydraulically-operated turrets were experimented with, but in the opinion of American armament experts these turrets could not be considered entirely satisfactory for mass production.

Radar controlled turrets had not been developed by the end of the war.

5.2. *Armour*

At the beginning of the war when the Japanese were using pilots fully trained in evasive manoeuvres, they considered that aircraft performance was more important than protection of the pilot, and hence fitted little or no armour plating.

As the war progressed they found difficulty in obtaining sufficient trained air crews, and they then provided armour protection for the pilot and air crew, but not for the engines or the engine oil cooling system, nor did they use self-sealing hose in the fuel system.

5.3. *Engines*

Japanese engine design and production was covered by another member of the Australian Mission. Hence the writer did not attempt a serious study of this subject.

The Japanese military considered it desirable to standardise on radial air-cooled engines. To the writer their designs appeared to be sound, but not brilliant, and in general similar to American designs. Their accessories, like some of our own, appeared to be on the heavy side for the work which was required of them.

5.4. *Engine Mountings*

In general engine mountings were made from chromium molybdenum steel tube arc or oxy flame welded, but towards the end of the war mountings were being produced in straight carbon steel due to shortages of alloy steels.

Japanese engine mounting designs were satisfactory for engines up to about 1,500 h.p., but on larger engines they experienced vibration troubles which were never solved.

5.5. *Electrical Systems*

Twelve volt two-wire systems were in more general use than twelve volt single-wire ground return systems. Some aircraft used 24-volt two-wire systems, and it was intended to standardise on this system. Batteries were of the acid lead type and most of the cable used had a natural rubber dielectric. It is believed that some polyvinyl chloride dielectrics were used, and also a dielectric material called Amiran which is similar to nylon.

5.6. *Fuels*

According to the Production Manager of the Mitsubishi Aero-engine Factory at Kyoto, the Japanese were unable to obtain an octane rating in excess of 78 with their own fuels, and not more than 92 by blending with imported fuel. Naturally, this seriously retarded engine design and hence aircraft performance.

5.7. *Gas Turbines*

Strictly speaking, this subject is outside the field to be covered by the writer, but the following remarks are considered necessary to explain why no comments on high speed jet airframes appear in these notes.

American technicians have studied this subject in detail and have formed the opinion that up to 1944 Japanese development in jet engines was non-existent. After 1944 until the end of the war the Japanese concentrated on making frantic efforts to copy the best from German designs, and showed no originality.

(When reading translations of Japanese reports on gas turbines it is important to note that their terminology differs from ours; they refer to any power plant using jet reaction as a " rocket "; this includes turbo-jets as well as solid and liquid-fuel rockets. They use the term " fuel injection " to describe all fuel systems which inject fuel by any means.)

5.8. *Instruments*

In general, the Japanese instruments are copies of American instruments, but inferior as the result of poor workmanship, and use of unsuitable alternative materials. The following list gives an indication of the types of instruments in general use:—

INSTRUMENT	TYPE
Altimeter	Aneroid
Tachometer ...	Dynamo or mechanical
Oil pressure ...	Burdan Tube
Fuel pressure ...	Burdan Tube
Oil temperature	Wheatstone Bridge
Cylinder temperature	Thermocouple
Fuel Gauge ...	Slip motor or D.C. Selsyn

Instrument panels were manufactured from wood or aluminium, and shock-mounted with Lord-type rubber mountings.

Up to 1940 indirect illumination of instrument panels by several lamps was used and after that date, ultra violet illumination.

In general automatic pilots were based on the American Sperry design, but an electric automatic pilot has been developed and used successfully on small aircraft. No electric automatic pilot was in production for larger aircraft.

5.9. *Power Operation of Landing Gear and Flaps*

In general, the Japanese used hydraulic power for operating landing gear and flaps. An operating pressure of 80 kg./sq. cms. (1,140 p.s.i.) was considered to be the most suitable pressure, although some Army aircraft had used 60 kg./sq. cms. (850 p.s.i.). The Navy had experimented with 120 kg./sq. cms. (1,700 p.s.i.) with some success, but Commander Otsuki, Chief Technical Officer at Yokosuka Naval Arsenal told the writer that considerable difficulty had been experienced in obtaining satisfactory oil sealing at these higher pressures.

Low and medium pressure hydraulic pipe lines were of aluminium alloy, and steel was used for the high pressure lines. Mineral base oils were standardised on. Pressure reducing valves were used to reduce the oil pressure to approximately 35 kg./sq. cms. (50 p.s.i.) for brake systems

In the early stages of the war both open return and closed return circuits were used, but recent designs all had closed return circuits with pressure accumulators.

5.10. *Pressure Cabins*

The greater part of the research work on this subject was undertaken by the Japanese Army Air Corps, which is an exception to the general rule that the Navy technicians were more progressive than the Army. As far as is known only one type of pressure aircraft was actually built. Pressure tightness of joints was obtained by double rows of rivets pitched four to six times their diameter and " Neoprene " was used to seal the joint.

In a discussion with the designer of this aircraft the writer confirmed an opinion already expressed by American technicians, namely, that the design had failed because insufficient attention had been paid to the development of suitable compressors and air cleaners.

5.11. *Primary Structures*

It has already been stated that a detailed investigation of specific types was impracticable. In general, Japanese designs for primary structures such as fuselages, wings and undercarriages are orthodox, except that greater attention is paid to reducing

unnecessary weight than is usual in our own designs. This reduction in weight is frequently obtained at the expense of simplified production.

5.12. *Propellers*

In the opinion of American technicians the Japanese propeller theory was at least five years behind current thinking in America. This is considered to have been one of the major limiting factors in the performance of Japanese aircraft.

5.13. *Protective Treatments*

The following protective treatments were used in Japanese aircraft:—

(a) Aluminium Alloys.

Clad sheets, pipes, castings and fuel tanks received no treatment. Unclad sheets and machined parts were sometimes anodised but generally two coats of paint was considered adequate.

(b) Steel.

Cadmium plating was in general use, but towards the end of the war zinc or tin platings were used.

(c) Magnesium Alloys.

As this alloy was in short supply it was not used very extensively, but when it was used it was generally in the form of castings which were treated by flash dichromate before machining to prevent deterioration in store, and selenious acid or dichromate treatment followed by a coat of paint after machining.

(d) Timber.

Wooden wing surfaces were treated as follows:—

(i) The veneer was rubbed with a paste of clay and water to fill the pores.

(ii) Two or three coats of clear cellulose acetate dope were applied.

(iii) A fabric cover of flax or ramie plain woven cloth.

(iv) Two or more coats of clear nitro cellulose lacquer sanded smooth.

(v) One coat of pigmented oil base paint.

5.14. *Radar and Radio*

Up to the end of the war the standard radio transmitter and receiver in mass production had a power output of 30 watts, a frequency of 30 to 50 mcs. and weighed about 33 lb.

The Japanese purchased a set of German Lorenz blind landing equipment which they endeavoured to improve on, without success.

Some experimental installations of two meter I.F.F. had been produced but, in the opinion of American technicians, the Japanese are considered to be far behind the Allies in this field.

5.15. *Self-Sealing Fuel Tanks*

The self-sealing tanks used by the Japanese had an aluminium alloy shell covered with four layers of natural rubber. These tanks had not proved satisfactory in service and a later development was produced but did not actually get into combat, so that its effectiveness was not assessed. The construction of this tank was as follows:—

A non-metallic fuel cell was suspended inside a .039 inch duralumin outer case, so that there was a 0.394 inch air space between the case and the fuel cell.

The inner layer of the fuel cell was 0.12 inches polyvinyl alcohol resin over which was placed two layers of natural rubber, and an outer layer of tyre chord impregnated with natural rubber.

5.16. *Standard Data Sheets*

The writer obtained a set of standard data sheets, as issued to Japanese design offices, from the Chief Designer of the Kawasaki Aircraft Co. In general these data sheets are similar to, and in many cases identical with, data sheets used by the Allies.

5.17. *Synthetic Rubber*

The Japanese used Buna "N" with a Shore hardness of 85 for hydraulic packings, and for fuel and oil hose they used Buna "N" or "Neoprene" (Chloroprene).

A typical assembly of sealing glands for an undercarriage oleo leg would consist of eight vee-shaped rings; four Buna "N" of 85 Shore hardness alternating with four tanned chrome leather rings of the same cross section, and metal collars used as retaining rings at each end of the gland assembly.

5.18. *Timber and Glues*

Towards the end of the war the Japanese were forced to produce many portions of their aircraft in wood instead of metal, due to serious shortages of the latter material. No fungicides were used, and the following table gives an indication of the timbers and glues in general use.

Component	Glue
(a) Wing Spars	
Impregnated hardwood (birch or beech)	Phenol aldehyde resin
Laminated wood (cypress)	Tego film
(b) Fuselage Longerons	
Japanese cypress or spruce	Urea aldehyde resin
Laminated cypress	Tego film
(c) Skinning	
Birch veneer	Milk casein or Tego film
Cypress veneer	Milk casein or urea aldehyde resin
(d) Propellers	
Impregnated birch and beech hardwood for the blade roots	Phenol aldehyde resin
Japanese spruce for blades.	Urea aldehyde resin

5.19. *Tricycle Landing Gear*

These landing gears were not out of the experimental stages.

5.20. *Tropicalisation and Winterisation*

At first no special precautions were taken for aircraft operating in the tropics and, as might be expected, trouble was experienced with dust and fuel vapour locks. To overcome these difficulties the Japanese developed an air cleaner based on a design which they found on a captured U.S. "Avenger," and lowered the volatility of their fuel.

(Normally the Japanese used 92 octane fuel or less, and although some research had been carried out on 100 octane fuel no large quantities were available.)

For aircraft operating in cold climates special shutters were used on oil coolers and radiators. Also, large drain taps for quick drainage of oil and coolant were fitted.

Oil dilution systems were in the experimental stage, and wing leading-edge rubber de-icing boots were of an unsatisfactory design and gave trouble in service.

5.21. *Undercarriages*

Most undercarriage oleo legs were of the simple straight leg type, and varying oil orifices were not considered necessary. In the opinion of American technicians who have studied this subject in detail, these oleo legs compared quite favourably with American standards. In only one aircraft had a hydraulic shimmy damper been used, and that was a direct copy of the damper used in Douglas aircraft.

Both up and down undercarriage locks of conventional design were in general use. Pre-rotation of wheels had been considered, but not tested.

6. THE VALUE OF JAPANESE RESEARCH TO THE AIRCRAFT INDUSTRY

Undoubtedly the most astonishing characteristic of Japanese Research Establishments

is their size. There are several research stations but the three most important are:—

(i) The Army Research Establishment at Tachikawa.
(ii) The Naval Research Establishment at Yokosuka.
(iii) The Research Institute at the Tokyo Imperial University.

All three of these establishments are fully, almost lavishly, equipped with facilities for all branches of aeronautical research, and between the three of them there are twenty to thirty wind tunnels, varying from a medium size full scale tunnel to a high speed supersonic tunnel. The general impression gained is that they have more equipment than they know what to do with.

The majority of the buildings had been built ten to fifteen years ago, and, as they were on a scale far more extensive than would be required for normal peace-time activities, it is assumed that the Japanese Government had been preparing for war for many years, and were only awaiting a favourable opportunity.

Of the two military research establishments, namely, the Army establishment at Tachikawa and the Naval establishment at Yokosuka, there is little doubt that the latter was the more efficient. In fact, the Army and civilian designers used the Navy's design requirements in the stress analysis of their aircraft; this was about the only thing that the two branches of the Japanese Air Forces had in common, the lack of co-operation between them amounting to a definite mutual distrust.

The distrust which the two branches of the Japanese Air Forces had for each other was increased when either one of them had to work in conjunction with research personnel. The military seldom gave a self-contained problem to any one research establishment, but split it up among two or three different establishments and discouraged discussions between the various research workers engaged on the problem.

There appeared to be a definite lack of co-ordinated executive control, both within any one laboratory and between different laboratories, which resulted in misdirected effort and inefficiency. An executive research committee did exist, but it was ineffective. This does not mean that the Japanese are incapable of undertaking useful research work; on the contrary, some Japanese did very valuable work but they definitely lacked co-ordination, and this had a serious effect on their value to the aircraft industry.

Regarding the nature of the research work being undertaken. In the mechanical test laboratories there were numbers of compression test specimens of theoretical cylindrical fuselage sections, with various combinations of formers and stringers; all very similar to work which had been undertaken in England and America some years ago.

Their equipment for testing completed wings varied from the shot bag technique to the use of hydraulic rams, capable of applying repeated loading with about a one minute cycle. The hydraulic rams used in one laboratory had integral electric motors, which automatically rotated the jack rams, and hence reduced errors resulting from friction losses during loading.

The Naval Research Establishment at Yokosuka had two pieces of research equipment which appeared to the writer to be of a novel nature. The first was an undercarriage test rig which had been built in the following manner.—

(a) A large wedge-shaped ramp capable of having the angle of the slanting side varied by hydraulic jacking had been constructed from steel girders.
(b) A trolley ran on rails along the slanting side of the ramp, and to this trolley the test undercarriage was attached.
(c) For testing, the undercarriage and trolley were drawn up the ramp, released, and allowed to run freely down until the undercarriage made contact with the ground.

(d) The required ground reaction and impact velocity was obtained by making suitable adjustments to the angle of the ramp, the ballast weights on the trolley, and the distance which the undercarriage and trolley were drawn up the ramp before release.

This rig was only just being completed at the end of the war, and had never actually been used.

The second test rig, of a type which the writer had not previously seen in England or America, was a fuel tank fatigue testing machine. The rig consisted of a horizontal table on which was mounted the tank to be tested. The table was vibrated in two mutually perpendicular directions on a horizontal plane, by means of variable speed electric motors and eccentric drives; the period of vibration being adjusted until resonance occurred in one or more of the tank panels. Commander Otsuki, the Technical Officer-in-Charge of this establishment, stated that all tanks tested had failed, and that he considered the test to be too severe.

7. PRODUCTION OF JAPANESE AIRCRAFT

The writer cannot claim to be a production expert, but believes that it did not require the experience of an expert to come to the conclusion that the Japanese were considerably behind the Allies in production methods. One of the most experienced Japanese engineers interrogated, a man with 22 years of aircraft experience and an undoubtedly wide knowledge, estimated that five Japanese workers were required for every one American or German worker. Nothing which the writer saw during his inspection of what remained of the Japanese Aircraft Industry led him to doubt the accuracy of this statement.

The general standard of finish on machined and pressed detail parts is quite equal to our own, and their standard of flush riveting and skinning, particularly on thin gauge sheet, is excellent. However, it is difficult to understand how they could keep accurate stores records, as detail parts were frequently stacked in open racks and it appeared that if a fitter damaged a part he could throw it away and use another.

One basic difference between Japanese practice and our own is that Japanese military research establishments frequently carried out complete detail design of new aircraft, and actually manufactured about 50 to 100 aircraft before handing the designs over to civilian factories for mass production. For example, the Naval Arsenal at Yokosuka, in addition to having about ten wind tunnels and associated research equipment, had a total staff of 15,000 to 20,000 and 25 miles of underground tunnels, used extensively on routine manufacture and assembly of aircraft.

8. INSPECTION METHODS

In theory Japanese inspection methods were similar to our own, except that raw materials, after release from the manufacturer by Army or Navy inspectors, were not normally check-tested by the aircraft constructors before release from their quarantine stores to production. Japanese designers considered that all important parts should be hardness-tested after heat treatment; that welded primary structures should be magnaflux tested; and casting radiologically examined. An inspection of the remains of the Japanese aircraft factories gives the impression that factory inspection was not as rigid as the designers would have wished.

The Japanese had a firm's inspection staff, and Navy or Army resident inspectors, in all factories; the firm's staff generally represented 5 to 10 per cent of the total physical staff.

9. ORGANISATION AND PERSONNEL

It has already been stated that Japanese research establishments were not of as much value to the aircraft industry as might have

been expected, due not to lack of equipment or staff, but to lack of organisation. In an attempt to make good this deficiency the Japanese designers frequently built up large research staffs within their own departments. An example of such an organisation is given in Appendix I, Part (b).

It appears that factory workers were not normally exempt from military call up, and the Manager of one of the Mitsubishi factories stated that towards the end of the war he had as many as 4,000 male and female students, average age about 16 years, sent to his factory to replace trained personnel. The same factory manager said that to have an effective staff of 7,000 employees he had to retain on his books about 9,300, as at any one time about 2,300 workers were absent because of sickness or pretended sickness.

Working hours were long, and a seven day week with two Sundays free per month was standard for both salaried and physical staff. Design staffs worked a ten-hour day and were frequently required to live on the factory premises. Physical staff generally worked a day shift of eleven hours and a night shift of eight hours.

Female labour was about 30 per cent on light work at the start of the war, and increased to about 60 per cent towards the end of the war. Female labour was not satisfactory on the night shift. Executives considered that production would have been increased by reducing working hours, but this view was not shared by the military. Working conditions were inferior to our own, but not nearly as poor as the writer had expected, and he was surprised to note that an Imperial decree required employers to give six monthly medical and dental attention to their staffs.

10. ACCIDENT STATISTICS

The following table indicates clearly that the Japanese were unable to give adequate training to their air crews towards the end of the war:—

Year	Flying Hours per Accident	Personnel Killed
1939	1,000	62
1941	1,050	88
1943	875	480
1944	750	700
1945	650	700

It has been estimated that approximately 80 per cent of accidents were due to personal causes such as carelessness, inadequate training, faulty maintenance and errors of judgment. Approximately 16 per cent were due to faulty design and/or defective materials.

11. TEST FLYING

There was no special flying training for test pilots and it was not considered necessary for them to have a technical background; they were, however, chosen from military pilots with good flying records. An interesting indication of the effectiveness of Japanese test pilots was obtained from the Chief Designer of the Nakajima Aircraft Co., who stated that out of eighteen test pilots at that company he could only rely on three to give him real assistance in prototype flying.

The four designers with whom the writer made personal contact did not themselves fly, although Mr. Horikoshi, designer of the " Zero " fighter, stated that he accompanied the test pilot on all preliminary tests except the first.

12. CONCLUSIONS

The conclusions formed by the writer, after reviewing available technical data at the Tokyo Headquarters of the American Air Technical Intelligence Group, interrogating typical Japanese aircraft technicians and visiting as many aircraft establishments as was practicable, can be summarised conveniently under the following headings:—

(a) The value of Japanese research to Japanese aircraft designers.
(b) The standard of design of Japanese aircraft.
(c) The production of Japanese aircraft.

(*a*) *The Value of Japanese Research to Japanese Aircraft Designers*

(i) Japanese research facilities should have been adequate to meet any demands which the aircraft designers might reasonably have been expected to make during the war years.

(ii) Japanese designers did not receive as much assistance from research establishments as they could have, and frequently built up excessively large private research departments.

(iii) It is considered that Japanese research establishments failed to be of optimum value to the aircraft industry because:—

(a) There was no virile co-ordinating committee to ensure that available facilities were used to the best advantage.

(b) There was a definite lack of mutual understanding and co-operation between the air forces and the research organisations.

(*b*) *The Standard of Design of Japanese Aircraft*

Immediately prior to the war Japanese aircraft designs compared favourably with our own, as most Japanese designers had a sound aeronautical training, generally obtained in America, and were able to use American, English and German technical reports to good advantage. With the declaration of war supplies of English and American reports ceased, and the Germans did not help the Japanese as much as might have been expected—due, presumably, to lack of mutual trust between the two countries. Thus the Japanese were forced to rely on their own Government research establishments to enable them to keep pace with Allied developments, and these establishments failed, not because of inadequate facilities, but lack of organisation.

The Japanese design firms built up large research departments, but were still unable to keep pace with Allied progress. It is the writer's opinion that, at the end of the war, Japanese aircraft designs were inferior to those of the Allies; this was particularly noticeable in such items as power-operated gun turrets, radar and, at a later date, gas turbines.

(*c*) *The Production of Japanese Aircraft*

(i) The Japanese custom of permitting Government research departments to undertake detail design and limited production of new aircraft types is novel, but of doubtful merit.

(ii) Japanese production methods are inferior to those used by the Allies, and the output which can be expected from the average Japanese physical worker is about one-fifth of the output obtained from an Allied worker.

(iii) The standard of Japanese workmanship compares favourably with our own.

(iv) Working conditions in Japanese aircraft factories were reasonably good, in fact, considerably better than standard of living of the majority of the Japanese people.

APPENDIX I.

(a) Summary of metallic materials used in Japanese aircraft, together with their chemical composition, recommended heat treatments and physical properties.

(b) Organisation charts of a typical Japanese aircraft design department.

Nº	NAME	C %	HEAT TREATMENT °C AIR COOL NORMALIZING	°C HEARTH COOL ANNEALING	WATER (AIR) COOL HARDENING	°C TEMPER	SPEC.	YIELD PT. kg/mm²	TENS STRENGTH kg/mm²	ELONG. %	CONTRACTION %	SHOCK VALUE mkg/cm²	HARDNESS B.H.N.
1.	EXTRA-MILD STEEL	<0·15	880-940	ABOUT 880	—	—	NORMALIZING	27-29	36-42	>30	>65	—	82-100
2.	SEMI-MILD STEEL	0·20-0·30	840-900	" 840	—	—	DITTO	>28	>45	>28	>45	>9	110-160
3.	MILD STEEL.	0·30-0·40	830-880	" 830	830-880	550-650	DITTO HARDEN & TEMP	>31 >40	>50 >50	>24 >24	>45 >55	>6 >10	120-170 170-230
4.	SEMI-HARD STEEL	0·40-0·50	820-870	" 820	820-870	550-620	DITTO	>50	>70	>17	>45	>8	200-250
5.	HARD STEEL	0·50-0·60	800-850	" 800	800-850	DITTO	DITTO	>60	>80	>14	>35	>6	230-280

1. Carbon Steel.

Nº	NAME	TENS. STRENGTH	ELONG.	RADIUS OF BENDING AT AIR TEMP.
1.	EXTRA-MILD S. PLATE	34-41	>25	r = 0
2.	SEMI-MILD S. PLATE.	39-47	>21	1 TIMES PLATE THICKNESS.
3	SEMI-HARD S. PLATE.	44-50	>20	1·5 "
4	HARD STEEL PLATE	>50	>17	2 "

Nº	NAME	TENS. STRENGTH	ELONG.	HEAT TREATMENT
1.	EXTRA-MILD S. TUBE.	>30	>30	ANNEALING
2.	SEMI-MILD S. TUBE	{>46 >44	>30 —	300-450°C 30 MIN. 550-650°C ANNEALING.
3	SEMI-HARD S TUBE	{>55 >78	— —	250-450°C 30 MIN. 300-450 °C 30 MIN.
4	HARD STEEL TUBE.	>47	—	ANNEALING.

2. Plates and Tubes of Carbon Steel.

Nº	NAME	C %	Ni %	Cr %	Mo %	OTHER MAT	HEAT TREATMENT ANNEALING °C	HARDENING °C	TEMPERING °C	Y. P. kg/mm²	TENS S kg/mm²	ELONG %	CONTRACTION %	SHOCK VALUE mkg/cm²	HARDNESS B.H.N.
1	86 kg Ni-Cr STEEL	0·25-0·32	2·5-3·5	0·6-1·0	—	—	820 HEARTH	820-870 WATER OIL	550-620 WATER OIL	>70	>86	>18	>50	>12	250-302
2	94 kg Ni-Cr S.	0.-32 ~ 0·40	3·0-4·0	0·7-1·3	—	—	DITTO	820-880 OIL	560-640 WATER OIL	>78	>94	>15	>40	>9	270-321
3	100 kg Ni-Cr. S.	0·20-0·30	2·7-3·5	1·0-1·5	<0·6	W<1·0 Va<0·25	650 AIR	800-870 OIL	540-640 WATER OIL	—	>102	>17	>40	>8	230-341
4	115 kg Ni-Cr S.	0·17-0·30	3·8-4·5	1·0-1·7	<0·5	W<1·1 Va<0·5	DITTO.	800-870 AIR-OIL	100-200 AIR, OIL	>100	>115	>14	>40	>8	350
5	160 kg Ni-Cr S.	0·28-0·35	4·0-5·0	1·0-1·8	<0·6	W<1·0 Va<0·25	DITTO	800-880 AIR, OIL	100-200 AIR, OIL	—	>160	>7	>25	>4	>447
6	TAHARD STEEL	0·18-0·32	2·8-3·2	1·4-1·8	>0·3	Mn 0·7-1·1	650 HEARTH	830-880 AIR	ABOUT 150 OIL	—	183 170 >110	13 15 >17	37 46 >50	7 8 >11	ABOUT 500 465-475 ABOUT 363

3. Special Steel (1).

3. Special Steel (1).

Nº	NAME	C %	Cr %	Mo %	Va %	Si %	Mn %	HEAT TREATMENT			Y P kg/mm²	TENS S kg/mm²	ELONG %	CONT %	SHOCK VALUE mkg/cm²	HARDNESS B H N
								ANNEALING	HARDENING	TEMPERING						
1	Cr SEMI-HARD S.	0·4-0·5	1·5-2·0	-	-	-	-	800° HEARTH	800-850 OIL	620-700 OIL, AIR	>68	>80	>15	>50	>10	250-290
2	Cr. Va. STEEL	0·3	1·00	-	0·16	-	0·50	-	850 OIL	600-650 OIL	-	-	-	-	-	-
3	DITTO	0·3-0·55	0·9-1·5	-	0·15-0·25	-	<0·6	780 HEARTH	830-850 OIL	650 OIL	-	>95	>16	-	>8	270-320
4	Cr. Mo STEEL	0·25-0·35	0·8-1·5	0·15-0·30	-	-	-	800 DITTO	900 OIL, AIR	600-650 OIL, AIR	70	85	15	50	10	200-275
5	Si Mn STEEL	0·45-0·55	-	-	-	1·8-2·2	0·6-0·9	-	850 OIL	400-500	70-80	80-95	15-10	-	-	-
6	Cr Mn STEEL	0·23	0·9	0·20	-	0·15	1·4	-	DITTO	ABOUT 600	>70	>85	>16	>5	>12	241

4. Special Steel (2).

Nº	NAME	C %	Ni %	Cr %	Mo %	HEAT TREATMENT				SPECN	Y P kg/mm²	T S kg/mm²	ELONG %	CONT %	S. VALUE inkg/cm²	HARDNESS SHORE
						NORMALIZING	ANNEALING	1ST HARDENING	2ND HARDENING							
1	LOW CARBON C. S.	<0·2	-	-	-	870-920 AIR	ABOUT 870 HEARTH	870-920 OIL	750-800 WATER	-	>32	>50	>21	>50	>12	80-90
2	Cr C.S.	0·09-0·25	-	0·55-1·50	-	-	580-620 HEARTH	-	-	ANNEALING & CEMENTATION	30-98	50-139	6-30	30-70	-	-
3	LOW NICKEL C. S.	<0·15	2·0-3·5	<0·5	-	850-900 AIR	ABOUT 850 HEARTH	820-880 OIL	730-780 WATER	-	>50	>80	>15	>40	>9	75-85
4	HIGH NICKEL C. S.	<0·15	3·5-5·0	<0·5	-	830-900 AIR	ABOUT 830 HEARTH	880-900 OIL	730-800 WATER	-	>70	>90	>15	>45	>8	DITTO
5	LOW Ni - Cr C. S.	<0·2	3·0-3·5	0·9-1·1	-	-	DITTO	880-920 OIL	770-800 WATER	-	>80	>100	>12	>40	>6	DITTO
6	HIGH Ni - Cr C. S.	<0·2	4·0-5·0	0·7-1·2	<0·5	830-850 AIR	DITTO	880-920 OIL	750-820 WATER		>90	>110	>12	>45	>8	70-85
7	Cr - Mo C. S.	0·14-0·20	-	0·80-1·10	0·80-0·30	-	600	780-800	-	ANNEALING	-	70	-	-	-	BHN 200
										HARDENING - TEMPER	60	90-110	8	-	-	-
8		0·17-0·23	-	1·00-1·30	0·20-6·30	-	-	-	-	DITTO	-	75	-	-	-	B H N 215
											80	110-135	5	-	-	-

5. Surface Cementation Steel.

Nº	C %	Cr %	Al %	Mo %	HEAT TREATMENT		SPECN.	Y P kg/mm²	T.S. kg/mm²	ELONG. %	SHOCK VALUE mkg/cm²	HEAT TREATMENT	HARDNESS B H N
					HARDENING	TEMPERING							
1	0·35	1·69	1·06	0·4	950°-30MIN OIL COOL	700 - 60 MIN OIL COOL	NOT NITRI.	73	86	25	17	900 OIL COOLING - 700 OIL COOLING	260 - 270
							AFTER NITRI.	82	88	6·8	7·4	500 - 100 hrs NIT.	830 - 890
2	0·36	1·39	0·34	0·3	900°-30MIN OIL COOL	700 - 60 MIN OIL COOL	NOT NITRI.	74	84	26	19	850 OIL COOLING - 700 OIL COOLING	260
							AFTER NITRI.	76	86	18	5·4	500 - 100 hrs NIT.	630 - 660
3	0·4 - 0·5	1·5 - 1·8	1·0 - 1·3	< 0·5	820 - 88 OIL COOL	700° OIL COOL	NOT NITRI.	> 75	> 85	> 15	> 10	(SHORE - 100)	900 - 1050
4	0·44	2·29	-	-	850 OIL COOL	DITTO	NOT NITRI.	80	95	24	16	950 OIL COOLING - 700 OIL COOLING	280 - 300
							AFTER NITRI.	84	101	7·5	7·6	500 - 100 hrs NIT.	1100 - 1130

6. Case Hardening Steel (Nitriding).

Nº	NAME	C %	Ni %	Cr %	SPEC.	Y P kg/mm²	T S kg/mm²	ELONG %	CONT. %	SHOCK VALUE mkg/cm²	HARDNESS B.H.N.
1.	Ni-Cr STAINLESS	< 0·2	8	18	—	8·0	104	18	62	—	—
					1050°C AIRCOOL	28	68	60	62	13·8	175
2.	Ni Cr STAINLESS	0·02-0·15	7·10	12 - 20	—	—	—	—	—	—	—
3.	Cr DITTO	0·2-0·4	0·4-0·8 / 1·5 - 2·0	11-14	—	—	—	—	—	—	—
4.	Cr DITTO	0·08-0·12	—	13 - 15	550° C ANNEAL	70-80	80-90	10-12	—	—	—
5.	Cr DITTO	0·05-0·12	(< 20)	17-22	—	—	—	—	—	—	—
6.											

7. Stainless Steel.

7. Stainless Steel.

N°	NAME	C %	Mn %	Ni %	Cr %	Mo %	HEAT TREATMENT		SPEC.	Y.P kg/mm²	TENS. ST kg/mm²	ELONG %	CONT. %	TEST OF BENDING
							HARDENING	TEMPERING						
2	Ni ST.	0·42	1·65	25	–	–	950° HEARTH	–	HARDENING & TEMPERING	–	73	47	59	–
3	Cr-Mo ST. TUBE	0·25–0·35	<0·6	–	0·8–1·1	0·15–0·25	800° AIR	700 AIR	DITTO	>50	>63	>15	–	180° r=3t
4	Cr-ST. TUBE	<0·2	–	<2·0	10–15	–	900° OIL	650 HEARTH	DITTO	>40	>60	>25	>40	180°
1	Ni-Cr S. TUBE	<0·2	–	8	17–20	–	850° OIL	400–500	DITTO	134–163	145–175	11–13	32–36	–

8. Special Steel Tube.

N°	NAME	C %	Si %	Mn %	Cr %	Va %	HEAT TREATMENT		SPEC.	Y.P kg/mm²	T.S kg/mm²	ELONG CONT % %	SHEARING S kg/mm²	HARDNESS B H N
							HARDENING	TEMPERING						
1	Cr-Va STEEL	0·45–0·55	0·1–0·2	0·7–0·9	1·0–1·2	0·15–0·25	840–870 OIL	400–500 OIL	ANNEALING	–	69·8	23·3 –	–	180
									850°C OIL 400°C OIL	160	165	12·2 48·2	88·5	ABOUT 400
2	Si-Mn STEEL	0·7	1	1–1·2	0·5–1·0	–	–	–	HARDENING & ANNEALING	160–180	180–200	8·6 –	–	
3	Si-Mn-Cr STEEL	0·55–0·65	1·0–1·2	0·8–1·2	<0·5	W<0·6	820–850 OIL	ABOUT 500 AIR	DITTO	>90	>125	>11 >35	SHOCK VALUE >300	>362 f

9. Steel Wire.

N°	NAME	C %	Si %	Mn %	Cr %	HEAT TREATMENT		Y.P kg/mm²	TENS S kg/mm²	ELONG %	SPECIFICATION
						HARDENING	TEMPERING				
1	CARBON STEEL	0·6–1·2	–	–	–				>45	>10	ANNEALING
2	Si STEEL	0·45–0·65	0·4–2·2	0·3–0·9	–	830–880 OIL WAT	–	135	150	7	SPRING TREATING
3	Si-Cr STEEL	0·45	1·00	0·5	1·0	820 OIL WAT	450	85	98	12	HARDENING & TEMPERING
4	Si-Mn STEEL	0·2–0·3	1·0–2·8	1·0–1·5	–	–	–	–	>65	>12	ANNEALING
5	Low Mn STEEL	0·45–0·53	<0·30	1·7–2·0	–	820 OIL	450	–	115–135	–	–

10. Steel Plate for Spring.

N°	C %	Mn %	Si %	P %	S %	Cr %	Mo %	Va %	HEAT TREATMENT			SPEC.	Y.P kg/mm²	T.S. kg/mm²	ELONG. CONT % %	SHOCK VALUE mkg/cm²	HARDNESS B.H.N
									ANNEALING	HARDENING	TEMPERING						
1	0·90-1·10	0·30	0·30	0·025	0·025	1·2-1·4	—	—	— —	—	—						
2	0·90-1·20	"	"	"	"	1·2-1·8	<0·5	<0·35	800 -850·ASH	800-880 OIL	< 200	ANNEALING	< 45	< 70	>17 >30	>1·5	< 220

(AFTER HARDENING AND TEMPERING, B.H.N 600, ROCKWELL C63 SHORE 88)

11. Steel for Ball and Roller Bearing.

No.	NAME	C %	Mn %	W %	Cr %	Co %	Mo %	B FLUX DENSITY	H MAGNETIC INTENSITY	HARDENING °C
1	CARBON STEEL	0·8-1·20	0·3-0·8	—	—	—	—	7000-8500	45-60	700-750 WATER
2	W. STEEL	0·65-0·80	0·2-0·5	5-6	<1·0	—	—	9500-11000	55-70	800-860 "
3	Cr STEEL	0·9-1·2	0·3-0·5	—	1·5-3·0	—	—	"	53-65	790-810 OIL
4	LOW Co STEEL	DITTO	"	—	5-6	5-6	—	9000-9800	85-100	850-930 OIL
5	SEMI Co STEEL	DITTO	"		8-11	8-11	1·0-1·5	9000-9500	140-165	"
6	HIGH Co STEEL	DITTO	"		9-11	14-17	1·0-1·5	7500-9500	170-200	"
7	K S STEEL	0·8-1·05	0·3-0·8	5-9	1·5-5·0	30-40	< 4·5	8000-9000	200-250	950-1000 OIL
8	MK STEEL	<1·5		Ni=30·8	Al=12			9000-9400	120-430	

12. Magnetic Steel.

No.	NAME	Cu %	Mg %	Si %	Fe %	Al %		HEAT TREATMENT HARDENING	HEAT TREATMENT TEMPERING	YEILD PT. kg mm²	T.S. kg mm²	ELONG. %	HARDENING B.H.N.
1	Al-Cu-Mg (a) D	3·3-4·2	0·3-0·7	<0·5	<0·6	0·3-0·7		ABOUT 420°C AIR 490-520°C WATER	= AT AIR TEMP	— 22	>24 >38	>10 >14	>65 >90
	(b) SD, 24S	3·8-4·8	1·2-1·8	<0·5	<0·6	0·4-1·0		490-500° WATER	DITTO	26	>42	>12	>100
2	Al-Cu-Si 25S	4·0-5·0	—	0·5-1·2	<0·8	0·5-1·2		510-530°C WATER COOL	120-160°C 12-24 HRS	22	>38	>14	> 90
3	Al-Cu-Mg-Si HIROMINE	3·5-4·5	0·4-1·0	0·5-1·2	<0·6	0·4-1·0		490-520°C WATER COOL	150°-180°C 6-24 HRS	25	>43	>8	>120
4	Al-Mg-Si A·SIS, ARMINE	—	0·4-1·0	0·7-1·5	<0·6	0·15-0·7		530-560°C WATER COOL	150-170°C 12-24 HRS	—	>30	>10	>80

1. Forged High Strength Al-Alloys.

CLASSIFICATION	NAME	Cu %	Mg %	Mn %	Si %	Fe %	Al %	SPECIFICATION	HEAT TREATMENT HARDENING	HEAT TREATMENT TEMPERING	YEILD PT. kg mm²	T.S. kg mm²	ELONGATION %
Al-Cu-Mg	D	3·8-4·2	0·3-0·7	0·3-0·7	<0·5	<0·6		—	490-520° WATER	AT AIR TEMP.	22	>38	>15
								SDO (ANNEAL AFTER ROLL)	ANNEAL 350°C	—	—	<25	>10
Al-Cu-Mg	SD 24S	3·8-4·8	1·2-1·8	0·4-1·0	<0·5	<0·6		SDH (CORRECT AFTER H.T.)	490-500° WATER	AT AIR TEMP.	28	>43	>14
								SDR (ROLL AFTER H.T.	"	"	32	>44	>10
Al-Cu-Mg	C17S	3·8-4·2	0·3-0·7	0·3-0·7	0·8-1·2	<0·6			490-505°C WATER	155-165°C 10-20 HRS.	28	>43	>14

2. High Strength Al-Alloy Plate.

	NAME	Cu %	Mg %	Mn %	Si %	Fe %	Al %	SPEC.	HEAT TREATMENT ANNEALING	HEAT TREATMENT HARDENING	HEAT TREATMENT TEMPERING	YIELD PT. kg/m m²	T.S. kg/m m²	ELONG %
OUTER PANEL	SA3	—	0·4-1·0	1·0-2·0	<0·5	<0·6		SDCO	ABOUT 360°C	—	—	—	< 23	>10
INNER PANEL	SD, 24S	3·8-4·8	1·2-1·8	0·4-1·0	<0·5	<0·6		SDCH	—	490-500° WATER	AT AIR TEMP.	27	>42	>12
								SDCR	—	"	"	31	>43	>10

3. High Strength Al-Alloy Plates.

	NAME	Cu %	Mg %	Si %	Fe %	Ni %	Ti %	Al %	HEAT TREATMENT		T.S.	ELONG.	HARDNESS
									HARDENING	TEMPERING			
1. Al-Cu-Ni	(a) Y-ALLOY	3.5-4.5	1.0-2.0	<0.7	<0.7	1.5-2.5	<0.7		ANNEAL AT 350 -370°C AND AIR COOL. 490-510°C. WATER QUENCH	ROOM TEMP. TO 200°C	> 22 > 30	> 5 > 3	> 60 > 90
	(b) RR 56	2.0	0.8	0.7	1.4	1.3	0.1		OIL QUENCH 525-535°C. WATER OR OIL QUENCH	165-195°C FOR 10 hrs	> 40	> 10	> 120
	(c) RR 59	2.25	1.6	0.5	1.4	1.3	0.1		525-535°C WATER OR OIL QUENCH	165-195°C FOR 10 hrs	> 35	> 10	> 120
2. Al-Si	32S (SILTON)	0.5-1.3	0.8-1.5	11.5-13.5	<0.8	0.5-1.3	—		ABOUT 500°C WATER QUENCH	ABOUT 160°C FOR ABOUT 20 hrs	> 35	> 2	> 110

4. Forged Heat-Resisting Al-Alloy.

NAME	Cu %	Mg %	Mn %	Si %	Fe %	Al %	HEAT TREATMENT			TENS. ST kg/mm²	ELONG. %	SHEARING STR. kg/mm²
							ANNEALING	HARDENING	TEMPERING			
Al- RIVET	<0.2	-	-	<0.6	<0.6		ABOUT 350°C	—	—	>15	>2	>8
D- "	3.3-4.2	0.3-0.7	0.3-0.7	<0.5	<0.6		"	470-520°C WATER COOL	AT AIR TEMP.	>38	>16	>26
MILD D RIVET	2.0-3.0	0.2-0.6	<0.2	<0.6	<0.6		"	500-520°C WATER COOL	"	>26	>20	>18
Hy RIVET	-	4.5-5.5	-	<0.5	<0.5		ABOUT 400°C	—	—	>25	>25	>17

5. Al-Alloy Rivet.

	Cu	Mg	Mn	Si	Fe	Zn	Cr	REMARKS
F.S.D	1.5-2.5	1.5-2.5	0.3-0.8	<0.5	<0.6	5-7	0.1-0.4	
H.D	0.8	1.5-2.5	0.3-0.8	<0.5	<0.6	5.0-5.8	0.1-0.4	EASIER WORKABLE EXTRUDE spd = (5²10) x (S D)

A.B	0·8	1·3-2·3	0·3-0·8	20·5	20·6	3·0-3·8	0·1-0·4	EASIER WORKABLE EXTRUDE Spd. = $(5^2/10) \times (5/D)$							

SPECIFICATION	NAME	Cu(%)	Mg(%)	Mn(%)	Si(%)	Fe(%)	Ni(%)	Ti(%)	Al(%)	HEAT TREATMENT HARDENING	HEAT TREATMENT TEMPERING	T.S. Kg/mm²	E Long %	HARDNESS B.H.N.
Al-Cu	(a) Nº 195	4·0-5·0	-	-	< 1·2	< 1·0	-	-		ABOUT 510° WATER COOL	ABOUT 150° 10 HOURS	> 15 > 26	> 5 > 3	- > 70
	(b) ES	10·0-13·0	-	-	< 0·7	< 0·8	Sn < 1·0	-				> 18	> 1	-
	(c) No. 12	6·0-8·0	-	-	< 0·7	< 0·8	-	-		-	-	> 14	> 3	-
Al-Cu-Si	HSA	3·5-4·5	< 0·2	-	4·0-5·0	< 0·8	-	-		-	-	> 16	> 2	-
	LAUTERL	3·5-4·5	< 0·2	-			-			ABOUT 500° WATER COOL	ABOUT 150° C 12-20 HRS.	> 28	> 1	> 85
Al-Cu-Ni	(a) Y-ALLOY	3·5-4·5	1·0-2·0	-	< 0·8	< 0·8	1·5-2·5	-		500-520° AIR WATER OIL	ABOUT 200° C 10 HOURS	> 18 > 28	- -	> 65 > 95
	(b) RR-50	1·0-3·0	< 2·0	< 1·0	0·5-3·0	0·5-1·5	< 1·8	< 0·4		510 °C AIR COOL WATER	150°-175° 10-20 HRS.	> 18 > 20	> 3 > 2	> 70 > 75
	(c) RR-53	2·0-3·5	0·5-2·5	< 1·0	1·25	1·4	0·5-2·0	< 0·4		- ABOUT 520° WATER	155° 175° 10-20 HRS.	> 18 > 28	- -	> 75 130-150
Al-Si	(a) SILMIN	-	-	-	1·10-14·0	< 0·8	-	-		-	-	> 18	> 4	-
	(b) 26 G	-	0·3-0·8	0·3-0·8	8·0-10·0	< 0·8	-	-		-	-	> 18	> 3	-
	(b) γ-SILMIN	-	0·3-0·8	0·3-0·8	8·0-10·0	< 0·8	-	-		ABOUT 520° OIL COOL	ABOUT 170° 24 HOURS	> 25	> 2	> 85
	(c) Zn SILMIN	< 1·0	-	-	7·5-9·5	< 0·8	Zn 6·5-8·5	-		-	-	> 19	> 4	-
Al-Mg	(a) Hy, SA 2	-	0·1-0·5	0·1-0·5	< 0·6	< 0·6	-	-		ANNEALING ABOUT 400° C	-	> 18	> 5	> 70
	(b) PANTITAL	-	0·5-1·0	0·5-1·0	2·0-5·0	0·45	-	< 0·2		-	-	> 15	> 1·5	> 60
	(b) ANTI-CORODAL	-	0·5-1·0	0·5-1·0	2·0-5·0	0·45	-	< 0·2		520° 540° C WATER COOL	ABOUT 160° C 12 HOURS	> 25	> 1	> 90
	(c) Ti SEEWASSER (KS SEEWASSER)	-	2·0-2·5	1·0-2·0	-	-	-	0·2 OR (Sb 0·2)		-	-	> 16	> 2	> 60
	(c) Ti SEEWASSER (KS SEEWASSER)	-	2·0-2·5	1·0-2·0	-	-	-	0·2 OR (Sb 0·2)		ABOUT 570° C WATER COOL	ABOUT 160° C 10-12 HRS.	> 25	> 1	> 80

6. Al-Alloy Casting.

NAME	Al %	Mn %	Zn %	Mg %	STATE	T.S kg/mm²	ELONG. %
THE 1ST CASTING	3·5-6·5	0·1-0·5	2·5-3·5		CAST ONLY	>18	>5
THE 2ND CASTING	8·0-11·0	0·1-0·5	—		DITTO	>15	>1
	"	"	—		HARDENING & TEMPERING	>21	>1
DIE-CAST	8·0-11·0	0·1-0·5	0·2-1·0		DITTO	>15	>1

1. Magnesium-Alloy Casting.

NAME	Al %	Mn %	Zn %	Mg %	STATE	T.S. kg/mm²	ELONG. %
THE 1ST CLASS.	5·0-7·0	0·1-0·5	<1·5		SHEET	>25	>8
"	"	"	"		FORGING	>25	>8
"	"	"	"		EXTRUSION	>22	>8
THE 3RD CLASS	<0·5	0·5-2·5	<0·5		SHEET	>20	>3
"	"	"	"		FORGING	>20	>2
"	"	"	"		EXTRUSION	>18	>2

2. Magnesium-Alloy Sheet, Forging and Extrusion.

NAME	Cu %	Mn %	Fe %	Al %	Ni %	Sn %	Zn %	T.S. %	ELONG. %
NAVAL BRASS	61-66	–	–	–	–	0.75-1.5	–	>35	>20
HIGH STRENGTH BRASS (Mn BRONZE)	55-58	<4.5	<2.0	<2.0	<5.0	–		60-65	>18
	56-59	<1.5	–	–	<2.0	<1.5		50-55	>18

1. Special Brass.

	NAME	Sn %	P %	OTHER %	Cu %	T.S. kg/mm²	ELONG. %
1	ROLLED OR FORGED	5-7	<0.3	<0.4		>47	>15
	EXTRUDED.	8-9	<0.3	<0.4		>47	>15
2	ROLLED SHEET	5-8	<0.6	S<0.4		>65	>7
	EXTRUDED WIRE.	5-8	<0.6	S<0.4		>87	–
3	CAST	10-20	<1.0	<0.75		>15	>1.5

2. Phosphor Bronze.

NAME	Sn %	Sb %	Cu %	OTHER %	T.S. kg/mm²	ELONG. %	BEARING S kg/mm²	HARDNESS B.H.N
WHIT Nº II.	91.4	3.94	4.31	Pb 0.18, Ni 0.15	–	–	–	–
YAMATO METAL.	90	6.3	3.6	Pb<1.0, Ni<1.0 Ca<1.0, P<1.0	8.75	–	7.3	31
ISODA METAL.	93	3.5	3.5	Pb6	8	11.5	5.6	25
"	80	11	3	Pb6	9	–	7.4	32

3. Bearing Metal.

ENGINEERING DEPARTMENT

ORGANISATION CHART OF THE TACHIKAWA AIRCRAFT CO.

AUGUST 1945.

CHIEF ENGINEER

Total Staff 274

Office Management Staff 106

	Staff
Inquiries	9
Clerical	20
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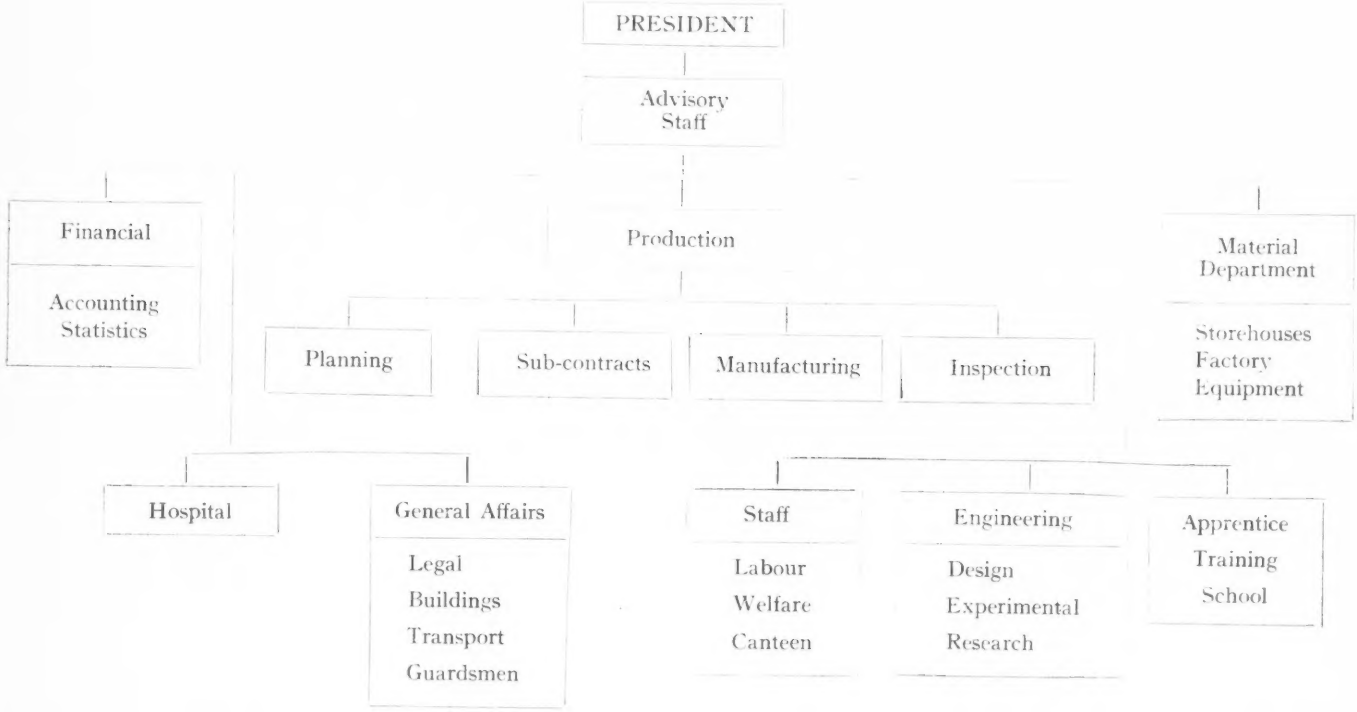
	Staff
Planning	8
Armament	5
Equipment	12

	Staff
Fuselage	14
Wing	6
Tail	4
Engine Mounting	16

Research Staff 103

	Staff
Aerodynamics	35
Structures	30
Electrical	8
Hydraulics	5
Equipment	17
Materials	8

ORGANISATION CHART
OF THE TACHIKAWA AIRCRAFT CO.
AUGUST, 1945.



REVIEWS

Fifty Years of Flying.

Griffith Brewer. Air League of the British Empire, London, 1946. 10/6 net.

Griffith Brewer first went up in the air—in a balloon—on the 9th May, 1891, so he is well within the truth of the title of his interesting book. He was the first Englishman to go up in an aeroplane, with Wilbur Wright himself as the pilot in 1908, and the author became a qualified pilot himself in 1914 at Dayton, Ohio, the Wrights' original flying field. In 1929 he took up flying for the second time and earned his English flying ticket on a Moth.

Since that first flight with Wilbur Wright in 1908 the author has been a close friend of Wilbur, until his death, and a close friend of Orville. No Englishman, indeed, has had closer and firmer relations with the two famous brothers, nor has anyone had such an intimate knowledge of their work and their personalities. For over thirty years he paid an annual visit to America, staying with the two brothers until Wilbur Wright's death, and afterwards with Orville Wright.

This is history told with first-hand knowledge, the history of ballooning from 1891 to 1908 and the coming of the aeroplane. Here one meets men who became famous; Patrick Alexander, who spent his fortune to help British aviation, Major Baden Powell, a past President of the Society and himself the second Englishman to go up in an aeroplane, Lord Brabazon of Tara, holder of No. 1 Pilot's certificate and a past President of the Society, Sir Francis McClean, who did so much to support the starting of aeroplane manufacture in England, Alec Ogilvie, the Short brothers, T. O. M. Sopwith, and many others.

Much of the second half of this book is concerned with the Wright machine, the fight the author made to vindicate their claim against the calumny of the Smithsonian Institution, culminating in a complete retraction in later years, and the founding of the now world-famous Wilbur Wright Memorial Lecture.

In June 1941 this astonishing grand young man of aviation, fifty years after he had first gone up into the air, flew the Atlantic. No one on board was half the author's age. The label on his luggage said: "Brewer, U.K.", the shortest address, as he remarks, "in all my experience of flying days."

This will be a book of historical importance in aviation literature. General Doolittle, when President of the Institute of Aeronautical Sciences, introduced Griffith Brewer on the occasion of his address to the Institute, in the following words: "His career in aviation spans more years than that of any man now living, and probably of any man in aeronautical history."

The reviewer hopes that Griffith Brewer will continue to add many more years to that span and create a record never likely to be beaten, for no man has done so much and so modestly for aviation than the author of "Fifty Years of Flying."

Proceedings of the Society for Experimental Stress Analysis.

Vol. III, No. 1. Addison-Wesley Press, Camb., Mass.

In this number the subject matter and its exposition amply maintain the high standard attained in earlier issues, and the Society for Experimental Stress Analysis is to be con-

gratulated on another most excellent publication. The field covered by the twelve papers is wider, if anything, than in previous numbers, containing papers on models and repeated loading as well as on strain gauging, photo elasticity, and stress crazing.

The papers on strain gauging comprise " Strain Gauge Technique," an excellent and well illustrated article on the practical details; " Precision Determination of Weight by Means of Bonded Strain Gauges," a description of how to weigh aircraft quickly and easily by using strain gauges; and the " Use of Strain Gauges on Wind Tunnel Models."

The papers on photo-elasticity and stress crazing are " Photo-elasticity as a Designer's Tool," indicating how this may be used in the *design* of specific parts; and " Dynamic Stress Analysis with Brittle Coatings," giving examples of this technique as applied to guns and undercarriages.

Of the remaining papers " A Study of Certain Mechanically-Induced Residual Stresses," which compares the residual stresses induced by shot blasting, surface rolling and cold bending; and " The Structural Analysis and Significance of Rivet Shear Tests," which contains some remarkable photographs of rivet failure, are of special interest.

The volume is well produced, and the many diagrams and photographs are excellent.

Jet Flight

John Grierson. Sampson Low, Marston & Co., Ltd., London, 1946. 18/- net.

Jets are in the fashion and the flow of books on Jet Propulsion will increase until it becomes a flood, which those who have so violently been propelled forward by their ideas hope will drown all other forms of prime mover. These enthusiasts are certain that the jet has turned the corner, but jets don't turn corners quite so easily as these enthusiasts may wish. There are many difficulties to be overcome before regular transport aircraft with jets are flying at great heights, great speeds and over long ranges with that reliability which their dying (*sic*) brother, the reciprocating engine, has given and is still giving. It is unwise to jettison tried and sound prime movers, eminently fitted for their jobs, for the sake of ringing out the old and bringing in the new. We are, unhappily, living in a world of change, but stability is born of steady motion.

This is not to suggest that jet propulsion will not be a great factor in the future of flight, but it is a hint that the enthusiast must now face up to the fact that he is over-selling his product, good as it undoubtedly is, and he may receive a set-back to his hopes.

The author has written a most interesting, detailed and inside account of the development of the jet, from the early work of Frank Whittle and the first flights of the E.28 to the gaining of the World's Speed Record of 606 m.p.h. in November 1945 by the Meteor IV. He writes well and with that knowledge of a pilot who himself took a considerable part in flying and testing jet propelled aircraft, and was the first pilot to make a cross-country flight. His own reports and those of the other pilots who did so much in those early days make not only fascinating reading, but are of great value for their detailed accounts of their test flights.

This is most certainly a book which aircraft designers should have on their shelves. The publishers are to be congratulated on the excellent printing and set up of a pre-war standard of production.

CORRESPONDENCE

PREFERRED NUMBERS

To the Editor. 15th May, 1946.

It is to be hoped that designers of components will give Mr. Conway's " Note on the Use of Preferred Number Series in Aircraft " (published in the May Journal) the attention which it merits. It is to be deplored that it is still normal for ranges of component sizes to be determined on a purely arbitrary basis, in spite of the fact that the Electric Lamp Industry gave the example to follow so very many years ago.

There is one aspect of the preferred number series, which follows automatically from the method of derivation, which I feel is not stressed sufficiently in the note. It is that, from any given preferred number series, a user can always select a size which is within a known tolerance of the value actually required. This tolerance is approximately 25 per cent. in the case of the 5-series, and 12 per cent. in the case of the 10-series (more exact values being 22.6 per cent. and 11.5 per cent. respectively).

A disadvantage of the 5- and 10-series is that for some purposes a 25 per cent. departure from the actual value required is greater than can be permitted, but 12 per cent. is closer than necessary. When preferred lists of Resistors were prepared for the use of the Radio Industry a few years ago it was desired that any value should be obtainable within 20 per cent. and in this case the 6-series was adopted, with the 12- and 24-series for use when greater accuracy was required.

The 6-series and 12-series are as follows:—

6-series	10		15		22		33		47		68		
12-series	10	12	15	18	22	27	33	39	47	56	68	82	

In view of the widespread knowledge of these particular series which has resulted from their use by the Radio Industry it would be necessary for them to be considered as possible alternatives to the 5-series and so forth, if it was decided to prepare a British Standard dealing with this subject.

It is to be regretted that Mr. Conway quoted, with apparent approval, the " rounded-off " table adopted by the A.S.A. This gives, in the 80-series, the following consecutive values: 28, 29, 30, 30.7, 31.5, 32.5. . . . It is difficult to reconcile these with a regular percentage increment, or, indeed, with any logical basis at all. The 40-series quoted has similar defects. If high order series are to be used it is essential that they be evaluated with appropriate accuracy if any advantages are to be gained from their use.

J. SPINKS, B.Sc.(Eng.), A.M.I.E.E.,
Associate Fellow.

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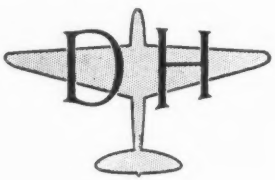

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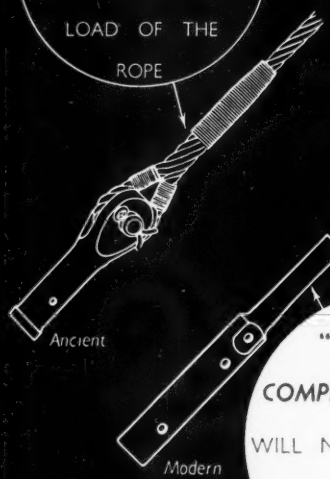

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